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ROBERTSON YOUNGQUIST, Editor

The JOURNAL OF THE AMERICAN ROCKET SOCIETY is devoted to disseminating information on the development of rocket and jet propulsion by printing original technical papers on jet propulsion, data on the latest experimental developments, historical notes, patent specifications, reviews of books and current literature, and news of the Society and individual members.

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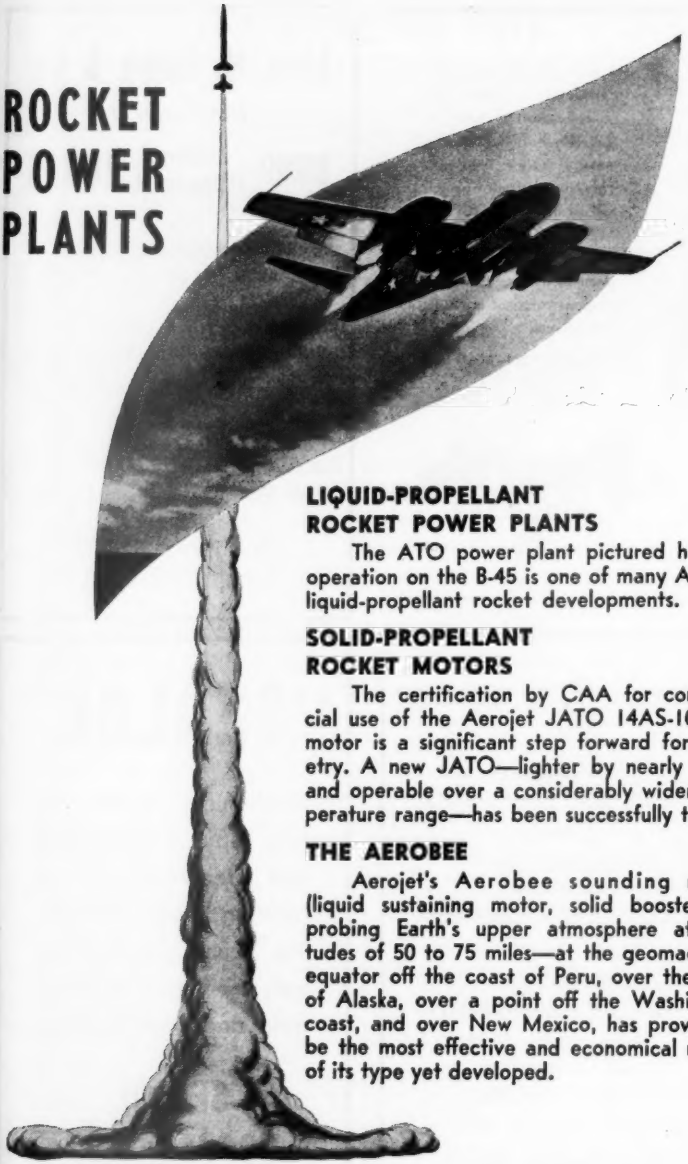
ROCKET POWER PLANTS



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JOURNAL OF THE
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Number 81 ROBERTSON YOUNGQUIST, *Editor* June, 1950

INSTRUCTION AND RESEARCH

at the Daniel and Florence Guggenheim Jet Propulsion Center

By Hsue-shen Tsien

Member ARS, Robert H. Goddard Professor, Daniel and Florence Guggenheim Jet Propulsion Center, California Institute of Technology, Pasadena, Calif.

IN "America Fledges Wings" R. M. Cleveland[1] stated: "Despite the importance of the role played by the Daniel Guggenheim Fund for the promotion of aeronautics as a herald, as an awakener, as a quickening spark in the manifold fields of practical aviation, its most important and probably most lasting contribution lay in its implementation and its creation of centers of research."

These centers of research are the well-known great schools of aeronautical engineering at the New York University, the Stanford University, the University of Michigan, the Massachusetts Institute of Technology, the California Institute of Technology, the University of Washington, and the Georgia School of Technology. It is a fact that a great majority of practicing aeronautical engineers today are either wholly educated in one of these centers or have had contact with one of these centers. Moreover, the strong influence of the Guggenheim Fund is not limited to this phase of aeronautical engineering. These centers of research contributed to a large extent to the fundamental knowledge of aeronautical science which forms the scientific basis of aeronautical engineering. Today we see an even more broadened effect exerted by the Guggenheim schools as men originally educated in these Guggenheim research centers establish new research laboratories and new schools of aeronautics in universities all over the world.

Jet-Propulsion Centers

The year 1930 marked the beginning of another phase of development instigated by the Guggenheim Fund. In that year Daniel Guggenheim

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[1] Numbers in parentheses refer to Bibliography on page 64.

made a special grant to the late Dr. Robert H. Goddard for liquid-propellant rocket research, to be carried on in Roswell, N. Mex. This work was continued under the Daniel and Florence Guggenheim Foundation until the death of Dr. Goddard. Research done by Dr. Goddard opened up the entirely new field of rocket engineering and heralded the dawn of the second epoch of aeronautics. This is the epoch of hyperaviation of flight with tremendous speeds at extremely high altitude. To propel a vehicle for hyperaviation, the conventional aircraft power plants are not adequate, and one must rely on radical propulsion systems such as rockets and ramjets. Dr. Goddard's work is among the first scientific experimentations in this field.

With this historical background of the Guggenheim Foundation in mind, it is then entirely fitting that the Foundation should decide during 1948 to establish two new centers of research to be called The Daniel and Florence Guggenheim Jet Propulsion Centers, one at Princeton University and the other at the California Institute of Technology. The purpose of the centers is threefold: (1) To train young engineers and scientists in the field of rocket and jet-propulsion technology on the postgraduate level, thus endeavoring to breed a new generation of pioneers to push the frontier of aviation to the next "higher" domain; (2) to instigate research and advanced thinking in rocket and jet propulsion, thus endeavoring to contribute to the basic knowledge necessary for the sound development of this new field; and (3) to promote peacetime commercial and scientific uses of rockets and jet propulsion. To carry out this program, there are the chairs of Robert H. Goddard professorships in honor of Dr. Goddard. Each Goddard professorship will be associated with a number of younger staff members and postgraduate fellows. The fellowships will be known as The Daniel and Florence Guggenheim Jet Propulsion Fellowships.

Instruction and Research of Jet Propulsion

Instruction and research in jet propulsion at the California Institute of Technology did not start however with the establishment of the Guggenheim Center; they started earlier.

During the academic year 1943–1944, the California Institute, at the request of the then Air Technical Service Command, Army Air Forces, initiated a course in rocket and jet propulsion limited to officer personnel assigned to the Institute for graduate study. Lectures were prepared by members of the Jet Propulsion Laboratory of the Institute and the GALCIT.[2] Several lectures on special topics were given by invited speakers. The course was planned by Theodore von Kármán, then Director of both the Jet Propulsion Laboratory and the GALCIT. The course covered in a comprehensive manner the basic principles of all jet-propulsion systems

[2] Guggenheim Aeronautical Laboratory, California Institute of Technology.

and the performance of jet-propelled vehicles. The lecture notes for the course, which was repeated during 1944–1945, were edited into the voluminous "Reference Text of Jet Propulsion" (2).

Research in rocket and jet propulsion at the California Institute started even earlier (3). The so-called GALCIT Rocket Research Project was initiated more or less informally in 1936. Early phases of the research were financed by a gift from Weld Arnold, now Colonel Arnold (a tremendous sum of approximately $1000!) This modest start led to a rapid growth under the exigency of war from demands of the AAF Materiel Command and the ASF Ordnance Department. The result is the Jet Propulsion Laboratory with a staff now numbering more than 575 persons and facilities valued at approximately $7,000,000.

Thus, it is fortunate that the newly established Guggenheim Jet Propulsion Center at the California Institute of Technology could obtain help and guidance from these two earlier developments in rocket and jet propulsion at the Institute, not to mention the inspiration it receives from the Guggenheim Graduate School of Aeronautics and the Guggenheim Aeronautical Laboratory. The center at the California Institute is a part of the Division of Engineering of the Institute. Its somewhat autonomous position is the result of the consideration that the solution of engineering problems in jet propulsion draws on knowledge and practice of older branches of engineering, particularly mechanical engineering and aeronautics. Thus, the program of instruction in jet propulsion should properly include material from both of these engineering fields. Furthermore, it is expected that, in general, students entering the course work in jet propulsion will have had their undergraduate preparation in mechanical engineering or aeronautics. Thus, the program of instruction in jet propulsion will have two separate options, allowing men from both aeronautics and mechanical engineering to follow their previous inclinations and developments. Both options lead to the degree of master of science upon the completion of the fifth-year program. For men in the aeronautics option, the degree of aeronautical engineer will be given upon the completion of a sixth-year program. Similarly, the degree of mechanical engineer will be given to men upon the completion of the sixth-year program of the mechanical-engineering option. More advanced study will lead to the degree of doctor of philosophy.

The actual courses of study in rocket and jet propulsion started September, 1949. Of course, the subjects of instruction and topics studied are in a state of flux. As experience is gained, there will be modifications in the materials covered and emphasis placed. It will take a number of years before the program of instruction can be stabilized.

The Guggenheim Jet Propulsion Center is not provided with large-scale research facilities as these are available at the Government-sponsored Jet Propulsion Laboratory at the Institute. It is expected that if basic re-

search in rocket and jet propulsion requiring expensive equipment coincides with the interest and program of the Jet Propulsion Laboratory, it could be carried out there.

Characteristics of Rocket and Jet-Propulsion Engineering

But what are the problems of basic research in rocket and jet propulsion? Before answering this question, it is profitable to keep in mind the salient features of rocket and jet-propulsion engineering. These are: (*a*) Short duration of operation of the power plant; and (*b*) extreme intensity of reaction in the motor. That the duration of operation of the power plant is short, stems from the fact of high specific consumption of propellant for the rocket engine. On the other hand, the dry weight of the rocket engine is much lower than that of other engines of equal output. Therefore the total installation weight of a rocket engine, which is the sum of dry weight and propellant consumed, can be lower than other power plants if the duration of operation is short (4). Furthermore, the specific consumption of rocket engines at all speeds and of ramjet engines at supersonic speeds in terms of lb per hr per lb of thrust is essentially independent of flight speed. Therefore the propulsive work done by the engine on the vehicle per lb of fuel or propellant consumed will be larger if the flight speed is larger. For this reason, it is advantageous to operate the rocket and the ramjet engines at large thrust and thus accelerate the vehicle to high speed. The large kinetic energy of the vehicle at the end of the "burning time" of the power plant is then utilized to achieve range by coasting. This form of dynamic trajectory is demonstrated to be superior to steady flight with long drawn-out operation of the rocket and the ramjet. Therefore all applications of these power plants will involve intensive but short duration operations of the engines.

Material Problems

The extreme intensity of reaction in the motor means high operating temperature. To find materials which can withstand high stresses at high temperature is then the main material problem in rocket and jet-propulsion engineering. However, the problem here is in one aspect characteristically different from the material problem in turbojet and gas-turbine design. This is the short operating time of the unit. For expendable units such as missiles, the operating time is generally of the order of minutes. Even for vehicles which are intended for repeated operation, it is still likely that the optimum performance is obtained by a design which requires replacing the high-temperature and highly stressed parts after each operation.

By adopting this concept of designing for minutes instead of designing for thousands of hours, as in the case of turbojets and gas turbines, the material will be stressed for ultimate strength and not for creep. This

difference is illustrated in Fig. 1 where the stresses are plotted against temperature. The lower curves are design curves for creep, and the upper curves are the ultimate stress, a stress which is practically independent of the rate of strain. For long operating time, the ultimate stress is not a design criterion, as the rate of strain near this stress is so large that the limiting strain will be reached long before the intended lifetime of the part, and the part will then fail. As seen from Fig. 1, if the part is designed to have a life of only a few minutes, it can be stressed six times higher. This is a tremendous possibility in design and occurs only in rocket and jet-propulsion engineering.

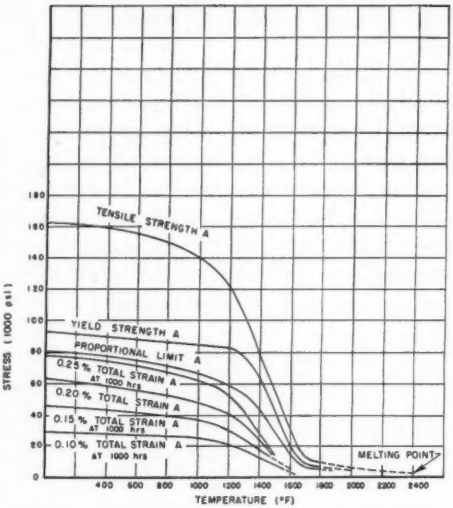

FIG. 1 STRENGTH VERSUS TEMPERATURE OF INCONEL X HEAT-TREATED 4 HR AT 2100 F, 24 HR AT 1550 F, AND 20 HR AT 1300 F

To explore this advantage leads however to complex problems in the stress and deflection analysis. The high rate of strain means constantly changing dimensions of the part, and its influence must be determined. The problem is not that of plasticity where the stress-strain relation is nonlinear, nor that of elasticity because now the material flows. In other words, the material must be considered as a viscoelastic medium. As a first approximation, the stress-strain relation can be still considered as linear. To be specific, let $\sigma_x, \sigma_y, \sigma_z, \tau_{xy}, \tau_{yz}, \tau_{zx}$ be the six stress components, $\epsilon_x, \epsilon_y, \epsilon_z, \gamma_{xy}, \gamma_{yz}, \gamma_{zx}$ be the six strain components. The stress-strain relation for isotropic viscoelastic media can be written as:

$$P\sigma_x = Q(\lambda e + 2\mu\epsilon_x)$$
$$P\sigma_y = Q(\lambda e + 2\mu\epsilon_y)$$
$$P\sigma_z = Q(\lambda e + 2\mu\epsilon_z)$$
$$P\tau_{xy} = Q\mu\gamma_{xy}, \qquad P\tau_{yz} = Q\mu\gamma_{yz}, \qquad P\tau_{zx} = Q\mu\gamma_{zx}$$

where λ and μ are constants and

$$e = \epsilon_x + \epsilon_y + \epsilon_z.$$

The operators P and Q are linear time operators defined as:

$$P = \frac{\partial^m}{\partial t^m} + a_{m-1}\frac{\partial^{m-1}}{\partial t^{m-1}} + \ldots + a_0$$

$$Q = \frac{\partial^n}{\partial t^n} + b_{n-1}\frac{\partial^{n-1}}{\partial t^{n-1}} + \ldots + b_0.$$

The a's and b's define the property of the material. They could be functions of time, but not functions of the space variables. Thus a material with changing properties, caused by the drift toward thermodynamic and chemical equilibrium, can be also represented by these operators.

An analysis (5, 6) of the mechanics of such materials reveals that if the load on the part is specified by a time factor $g(t)$, then the stress distribution at any instant can be calculated as if the material is purely elastic with the same instantaneous load. The deflection of the structure is, of course, different. But it is specified by a time factor $h(t)$ which is independent of the particular value and distribution of the load and is only dependent on $g(t)$ and is determined by

$$Qh(t) = Pg(t).$$

The $h(t)$ is thus a "universal" function in the sense that it is related only to $g(t)$ and the properties of the material. The other characteristics of the problem do not enter into its determination. In particular, the function $h(t)$ may be directly measured experimentally on a pure tension bar with the tension varied with time according to $g(t)$. This is then a considerable simplification of the mechanics of viscoelastic media and a useful tool in the application of the idea of design for short-time flow of material.

Heat Transfer

The extreme intensity of reaction in the rocket motor and in the combustion chambers of ramjet and pulsejet, and the high velocity of gas flow lead to a high rate of heat transfer to the walls, if the wall is to be kept at safe operating temperature. For instance, at the throat of a rocket nozzle, heat flux as high as 12 Btu per sec per sq in. has been observed. Changed into conventional units in other branches of engineering, this is more than 6 million Btu per hr per sq ft. To cope with this high heat flux, designers have been forced to extrapolate the empirical laws of heat transfer to a cooling liquid and to seek other unconventional methods, such as surface boiling heat transfer.

To absorb the high heat flux by circulating a cooling liquid in a duct surrounding the hot chamber, one must use large differences between the wall temperature of the cooling duct and the bulk temperature of the cooling liquid under turbulent-flow conditions. Here the problem is the lack of proper understanding of the basic mechanism. At present, the designer relies on empirical rules which are only safe to use within the range of variables of the test result. To extrapolate without the guidance of a sound understanding of the phenomena is not satisfactory. Of course the problem of turbulent heat transfer has been attacked successfully by O. Reynolds, L. Prandtl, G. I. Taylor, Theodore von Kármán, and others. But their work is based upon the assumption that the temperature difference between the wall and the bulk of the liquid is small so that the flow is essentially isothermal.

Turbulent flow in a duct, say a circular pipe, can be divided into three regions (Fig. 2): the turbulent central core, where Reynolds' turbulent shearing stress dominates the molecular or viscous shearing stress; the laminar layer next to the wall where the viscous shearing stress dominates the turbulent shearing stress; and the buffer layer where both shearing stresses are important. For the turbulent central core, which occupies most of the pipe, previous experiments with isothermal flow indicate that the flow in general, and the velocity profile in particular, are controlled by the shear stress τ at the boundary of the turbulent core, the density ρ of the liquid, and a linear dimension y_1. Since the boundary of the turbulent core is very close to the wall, τ is practically equal to the wall shearing stress τ_0. Together with ρ, τ_0 can define a velocity U_τ by

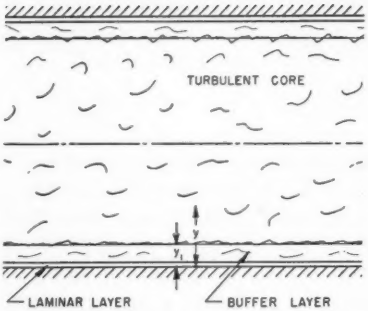

FIG. 2 THE THREE REGIONS WHICH OCCUR IN TURBULENT FLOW IN A DUCT

Turbulent core:

$$\frac{U}{U_\tau} = 13.94 + 55 \log_{10} \frac{y}{y_1}$$

Buffer layer and laminate layer:

$$\frac{\tau}{\rho} = \nu \frac{dU}{dy} - \overline{u'v'} + \frac{d\nu}{d\overline{T}} T' \frac{\overline{\partial v'}}{\partial x} + \frac{\partial u}{\partial y}$$

$$U_\tau = \sqrt{\frac{\tau_0}{\rho}}.$$

Then if U is the velocity at a point y from the wall, the nondimensional equation for the velocity profile must be

$$\frac{U}{U_\tau} = f(y/y_1).$$

Clearly, the only available linear dimension for flow near the boundary of the turbulent core is the distance of this boundary from the wall. Hence, for flow near the boundary of the turbulent core, y_1 must be the thickness of the laminar and buffer layer. From previous experiments with isothermal flow in smooth pipes (7), it is found that

$$\frac{U}{U_\tau} = 13.94 + 5.5 \log_{10} \frac{y}{y_1}.$$

Since temperature differences in the liquid will change only the viscosity, and according to experiments viscosity does not enter directly into the turbulent core flow, the velocity relation given above must also hold for nonisothermal flow.

The problem now is to determine the thickness y_1. This thickness will vary with the temperature conditions. The work of H. Reichardt (8) does not account for this variation and is therefore unsatisfactory. The main effect of higher temperature differences therefore is in the buffer layer

and the laminar layer. Here the variation of viscosity with temperature changes the flow. For instance, the effective shear stress τ is given by

$$\frac{\tau}{\rho} = \nu \frac{dU}{dy} - \overline{u'v'} + \frac{d\nu}{dT} \overline{T' \left(\frac{\partial v'}{\partial x} + \frac{\partial u'}{\partial y} \right)}$$

where ν is the kinematic viscosity, T the temperature, $u'v'$ the instantaneous turbulent velocities in the directions parallel to the wall and normal to the wall, and T' is the temperature fluctuation. The bar over the second and third terms means averaging with respect to time. The third term does not occur for isothermal flows. Its appearance and the appearance of the variable ν in the first term mean the effects of heat conduction and the effects of shear are now coupled. The solution is thus more difficult than the corresponding isothermal problem, but the difficulty is believed to be surmountable.

When the wall temperature is raised beyond that of the boiling of the liquid under prevailing pressure in the pipe, local vaporization takes place and bubbles are formed over the surface. But since the main bulk of the liquid is still at a temperature below the boiling point, these bubbles cannot grow indefinitely. In fact experiments by F. Kreith and M. Summerfield show that they contract again and have a life span of about $^1/_{100}$ sec. During its short life span, the bubble does not seem to move appreciably from the wall. The main consequence of the bubble formation and disappearance is then the strong agitation of the fluid near the wall. It is then understandable that the heat flux can be increased to many times that of the case without local boiling. This fact is shown clearly in Fig. 3, taken from the work of Kreith and Summerfield (9, 10). This means that a high rate of cooling can be achieved without high flow velocity in the cooling duct. The thus reduced pressure drop in the cooling duct will decrease the necessary pumping work of the coolant. Boiling heat transfer then can be used to good

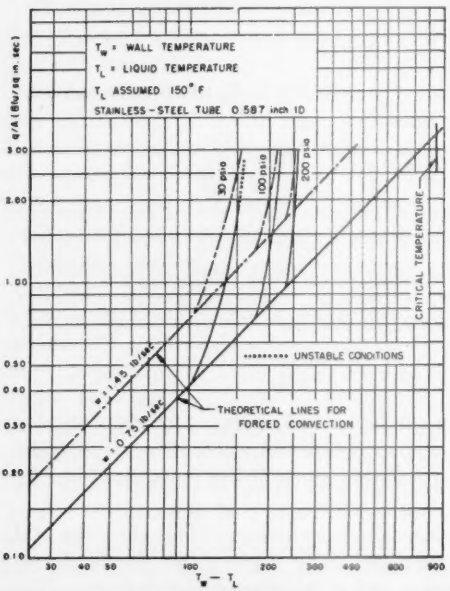

FIG. 3 COMPARISON OF FORCED CONVECTION HEAT TRANSFER WITH AND WITHOUT SURFACE BOILING

advantage for many designers. The problem for research here, of course, is a closer understanding of the turbulent agitation due to bubble formation and thus better correlation of tests for different liquids and different test conditions.

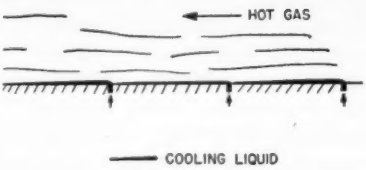

FIG. 4 FILM COOLING SHOWING DOWNSTREAM FLOW OF COOLING UNIT

If the wall temperature is increased beyond a critical value over the boiling point of the liquid, it has been found that a vapor envelope forms over the surface and the heat flux is reduced by the insulating effect of the stagnant vapor. Therefore, with specified pressure and flow velocity, there is definitely a maximum value of heat-flux density even with local boiling at the surface. If still high heat-flux density is desired, then other means of cooling have to be used. However even before reaching this intrinsic limit of boiling heat transfer, the wall temperature at the inside surface of the rocket motor may be too high for the material strength, due to the necessary temperature gradient through the wall for the heat flux. For instance, if the heat flux is 6 Btu per sec per sq in., the wall thickness is $^1/_{16}$ in., and the temperature at the cool side of the wall is 600 F, the temperature at the hot side of the wall will be 1950 F if stainless steel is used. This temperature is certainly too high for good strength. New powerful cooling methods for extremely high heat flux are sweat cooling and film cooling.

Film cooling (Fig. 4) is achieved by establishing a thin liquid film in contact with the hot gas over the surface to be cooled. Due to the shearing stress acting over the liquid-gas interface, the liquid flows in the downstream direction. Simultaneously, the heating of the film by the hot gas evaporates the liquid. It is seen that so long as there is a liquid film, the wall temperature is kept below the boiling point of the liquid. It is also seen that to protect the wall from the hot gas, liquid film has to be reestablished by injection through holes in the wall when the film from upstream injection is evaporated. Of course, the intervals of injection can be lengthened by injecting more liquid and establishing a thicker film each time. However, the difficulty here is the instability of the film against the turbulent flow in the gaseous boundary layer. The resultant partial breakaway of the liquid in the form of droplets constitutes a loss in effective cooling liquid. The problem here is then the determination of the relative cooling efficiency with respect to film thickness.

From experiments on one-phase turbulent boundary layer, it is found that the laminar sublayer thickness y^* is determined by

$$y^* = 5\frac{\nu}{U\sqrt{c_f/2}}$$

where ν is the kinematic viscosity of the fluid, U is the free stream velocity, and C_f the local friction coefficient. If this relation also holds for two-phase turbulent boundary layers such as exist in film cooling, y^* is the limiting film thickness for perfect efficiency. If the film thickness is larger than y^*, instability of the film and breakaway of droplets is likely to occur. It is then seen that there is an advantage in having a higher kinematic viscosity ν, as a thicker layer is then allowed. If $U = 1000$ ft per sec, $C_f = 0.004$, and $\nu = 0.319 \times 10^{-5}$ ft^2 per sec for water at 212 F, y^* is only 4.3×10^{-6} in. This result indicates that for theoretical maximum cooling efficiency, the film should be thin and re-established frequently along the wall. The limiting case is then sweat cooling where the coolant is forced through the porous wall, and injection and evaporation occur at the same time.

Sweat cooling is, however, not limited to the liquid coolant. The coolant may be gaseous. In fact, the most extensive experiments are made by P. Duwez and H. L. Wheeler (11) with gaseous coolants. It is shown by the above investigators however that the coolant cannot be allowed to evaporate in the porous wall, as then the flow is essentially not stable with wide fluctuations in the wall temperatures. Generally then, the most efficient sweat-cooling system with least expenditures of the coolant is one that evaporates the liquid coolant on the "outside" surface of the porous wall before entering the porous material, Fig. 5. In a sense, this system is a combination of boiling heat transfer and sweat cooling. No extensive experiment on this method of cooling has yet been made.

Combustion

It is evident that with either film cooling or sweat cooling, there is no limit to the temperature of the combustion gas that can be effectively handled. One need not have therefore any misgivings about high energy fuels and propellants for cooling difficulties. Furthermore, for rocket, ramjet, and pulsejet, there is no contact of the combustion gas with a delicate moving part such as turbine blades in a turbjoet; and the combustion gas can be corrosive and can contain finely divided solid particles. These factors practically remove all restriction in the choice of fuels and propellants. Such strange combinations as liquid hydrogen and liquid fluorine, and diborane (B_2H_6) and air are to be considered. To consider these combinations in combustion immediately raises two types of problems. The first type is problems in thermochemistry. What is the heat of combustion of these compounds? What are the thermodynamic properties of the

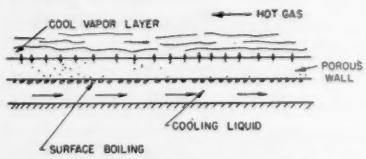

FIG. 5 SWEAT COOLING SHOWING EVAPORATION ON THE OUTSIDE OF POROUS WALL BEFORE LIQUID COOLANT ENTERS POROUS MATERIAL

combustion products? What are the equilibrium constants of the reactions? Information is particularly meager in connection with lithium, boron, fluorine, etc. The second type of problem concerns questions in chemical kinetics. How are the reactions of these combinations carried out? What are the individual component reactions? What are the rates of these component reactions? How can the reactions be speeded up or retarded?

A first application of this thermochemical and chemical-kinetic information to combustion problems is the calculation of the absolute flame speed. At present, it is generally accepted that the old thermal ignition theory of flame speed, due originally to Mallard and Le Chatelier, is unsatisfactory. Due to the complexity of the problem, different investigators have emphasized different aspects of the problem. No simple general theory has yet evolved. One basic assumption of these theories however, seems to need a clarification. This is the assumption that the rate of reaction calculated on the basis of homogeneous reaction can actually be used in determination of reaction in a flame front where the composition of the gas sometimes varies greatly in a distance of only a score of mean free paths of the molecules. In other words, the question is whether the nonuniformity involved can seriously change the reaction rate. What is needed then is a kinetic theory of nonuniform gas involving chemical reaction. For instance, to calculate the flame speed of a mixture of hydrogen gas and iodine vapor, one has to consider four species of molecules, H_2, I_2, HI, and $(HI)_2$ which is the activated complex of H_2 and I_2 after a successful collision. The Boltzmann differential-integral equation for the distribution functions of the molecules has to be solved by considering all possible collisions between these molecules. For some of the collisions the kinetic energy is not conserved.

More urgent problems of combustion in jet propulsion are those connected with fluid-mechanical aspects. These are the autoignition of liquid jets, the evaporation of liquid droplets, the mixing of gaseous components, the mutual influence of combustion and turbulence of flow, combustion in heterogeneous mixture, etc. For ramjets, the most perplexing problem today is the problem of flame stabilization. This is a problem which confronts all ramjet designers. Worse still, the mechanism of flame stabilization is not yet understood. As a result, the flame-holder design for the combustion chamber is always done by *ad hoc* experimentation. Clearly, there is a need for experiments with the simplest physical conditions so that the parameters can be accurately controlled. The work of A. S. Scurlock (12) with homogeneous gas stream and controlled turbulence in the initial stream is the most noteworthy effort in this direction. For a true understanding of the mechanism however, further detailed exploration of flow field is necessary.

It seems that one of the important aspects of flame stabilization is the

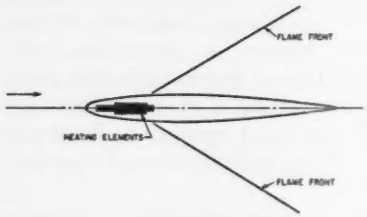

Figure 6

FIG. 6 FLAME HOLDER IN SHAPE OF STREAMLINED BODY

interaction of the flame front and the boundary layer. To test this concept, a flame holder in the shape of a streamlined body may be tried (Fig. 6). To start the combustion, the airfoil has to be first heated by, say, electric current to a high temperature. Once the flame is started, the airfoil aft of the flame is heated by the hot gas. Heat is then conducted through the body of the airfoil to the front section where heat is given up to the cold gas mixture through the boundary layer. The cold gas mixture in the boundary layer being heated by the body will increase the concentration of active carriers and finally ignite at the intersection of the flame front and the surface of the body. It is evident that such a flame holder without turbulent and eddying loss has many practical applications in spite of the difficulty in starting. In fact, by increasing the length of the airfoil in the hot gas, the temperature at the front part can be increased, and thus, the gas velocity can be raised without blowouts.

Performance of Rocket and Jet-Propelled Vehicles

The ultimate aim of all this basic research is of course to improve the performance of rocket and jet-propelled vehicles. However even when given the best power plant, the designer still has to determine the best way of using the power plant for optimum performance of the completed vehicle. For instance, what would be the optimum thrust programming for a sound rocket? What would be the gain possible by varying the thrust during ascent? Is this gain justified by the additional complication in the design? The basic variational problem of thrust programming was studied by G. Hamel (13). However, he made no detailed calculations to allow the designer to weigh the importance of different aspects of the problem.

But the fundamental question in the performance analysis is the trajectory, particularly long-range trajectory. Earlier in this discussion the reason for favoring the dynamic trajectory of varying velocity was given. But what particular dynamic trajectory? To avoid the penalty of high drag at high velocity in the dense atmosphere and yet to be able to accelerate the vehicle quickly, it is clear that the vehicle should be launched in vertical position. Performance of the vertical trajectory of a rocket is well known. But is rocket the only power plant capable of vertical trajectory? Certainly the ramjet, once boosted to a sufficiently high velocity, can also produce enough thrust to make an accelerated vertical flight. A ramjet would weigh more than a rocket per lb of thrust produced, but the

fuel consumption is much smaller. Preliminary estimate by L. H. Schindel (14) shows that the advantage of low fuel consumption overcomes the disadvantage of heavier dry weight. Therefore, there is a definite gain by substituting the lowest stage of a multistage rocket with a ramjet. Of course, to boost the ramjet to operating speed, it has to be operated as a ducted rocket in the first few seconds.

What can one say about the remainder of the trajectory? Since the high velocity of the vehicle is reached outside the atmosphere by vertical or near vertical ascent, the first part of the trajectory has to be liftless, and thus is elliptical. When the vehicle returns to the atmosphere at practically the same speed as it leaves the atmosphere, the lift of the body of the vehicle can be produced by putting the body into an angle of attack. The question here then is one of programming the angle of attack of the body so that maximum range is obtained. As an example of such a dynamic trajectory, the flight of a 3000-mile rocket vehicle is studied under the assumption of steady glide after the initial elliptical path. The average lift-drag ratio in glide is taken to be 4. The result of this analysis is as follows:

Length	78.9 ft
Maximum diameter of body	8.86 ft (Fig. 7)
Gross weight	96,500 lb
Fuel load	72,400 lb
Weight at end of burning	24,100 lb
Propellant loading fraction	0.750
Exhaust velocity	12,000 ft/sec
Propellant	Liquid O_2 liquid H_2; liquid F_2 liquid H_2
Maximum velocity	9140 mph
Range at conclusion of elliptic path	1200 miles
Range contributed by glide	1800 miles
Altitude at beginning of glide	27 miles
Landing speed	150 mph
Landing angle	20 degrees
Flight duration	Less than 1 hr

It is thus seen that the requirements of a transcontinental rocket liner is not at all beyond the grasp of present-day technology. The wings need not be large to achieve a reasonable landing speed, and the specifications on the structural weight are not impossible. When will such a rocket liner be realized? That is a difficult question to answer.

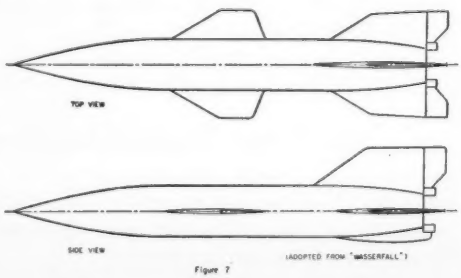

Figure 7

But one thing is certain: The basic research as outlined in this discussion will definitely hasten that day of long-range rocket travel.

Bibliography

1 "America Fledges Wings," by Reginald M. Cleveland, Pitman Publishing Corporation, New York, N. Y. (1942).

2 "Jet Propulsion," a reference text prepared by the staffs of the Guggenheim Aeronautical Laboratory and the Jet Propulsion Laboratory of the California Institute of Technology, Air Materiel Command (1946).

3 "Research and Development at the Jet Propulsion Laboratory, GALCIT," by R. Stanton, *Engineering and Science Monthly*, vol. 9, no. 7, 1946, pp. 5–14.

4 "A Guide to Aircraft Power-Plant Selection," by A. L. Lowell, *Aeronautical Engineering Review*, vol. 6, 1947, no. 4, pp. 22–25.

5 "Nonhomogeneous Stresses in Viscoelastic Media," by T. Alfrey, *Quarterly of Applied Mathematics*, vol. 2, 1944, pp. 113–119.

6 "A Generalization of Alfrey's Theorem for Viscoelastic Media," by H. S. Tsien, *Quarterly of Applied Mathematics*, vol. VIII, 1950, no. 1, pp. 104–106,

7 "Gesetzmässigkeiten der turbulenten Strömung in glatten Rohren," by J. Nikuradse, Verein deutscher Ingenieure, *Forschungsheft*, No. 356, 1932.

8 "Die Wärmeübertragung in turbulenten Reibungschichten," *Zeitschrift für Angewandte Mathematik und Mechanik*, vol. 20, 1940, pp. 297–328.

9 "Heat Transfer to Water at High Flux Densities With and Without Surface Boiling," by F. Kreith and M. Summerfield, *ASME Transactions*, vol. 71, 1949, pp. 805–815.

Also: "Investigation of Heat Transfer at High Heat-Flux Densities: Experimental Study With Water of Friction Drop and Forced Convection With and Without Surface Boiling in Tubes," Progress Report No. 4–68, Jet Propulsion Laboratory, California Institute of Technology, 1948.

10 "Heat Transfer from an Electrically Heated Tube to Aniline at High Heat Flux," by F. Kreith and M. Summerfield, Progress Report No. 4–88, Jet Propulsion Laboratory, California Institute of Technology, 1949.

11 "Experimental Study of Cooling by Injection of a Fluid through a Porous Material," by P. Duwez and H. L. Wheeler, *Journal of Aeronautical Sciences*, vol. 15, 1948, pp. 509–521.

12 "Flame Stabilization and Propagation in High-Velocity Gas Streams," by A. C. Skurlock, Meter Report No. 19, Massachusetts Institute of Technology, 1948.

13 "Uber eine mit dem Problem der Rakete zusammenhängende Aufgabe der Variationsrechnung," by G. Hamel, *Zeitschrift für angewandte Mathematik und Mechanik*, vol. 7, 1927, pp. 451–452.

14 "Application of Ramjet to High Altitude Sounding Vehicles," by L. H. Schindel, Master of Science Thesis, Department of Aeronautical Engineering, Massachusetts Institute of Technology, 1947.

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HEAT-TRANSFER PROBLEMS IN LIQUID-PROPELLANT ROCKET MOTORS

By Robert Gordon

Engineer, Aerojet Engineering Corporation, Azusa, Calif.

The present-day high-performance rocket motor, with its extremely high heat-flux rates and combustion temperatures, presents many new heat-transfer problems which are beyond the range of previous work and not adequately covered by existing theory. Some of the basic problems are discussed, and possible reasons for the inability of conventional formulas and theories to correlate actual rocket-motor heat-transfer data are given. Several techniques for handling high heat-flux rates and their applicability are mentioned.

THE major heat-transfer problem in rocket motors stems from the fact that the intensity of heat transfer is extremely high. The heat-flux rates encountered are of such magnitude that motor walls must be kept thin in order to prevent a large wall-temperature rise which will increase the operating temperature of the wall. Before examining this problem in detail, it may be worth while to establish a frame of reference for heat-flux rates.

The usual units of heat flux are Btu per hr per sq ft. In rocket-motor work it is more convenient to use the combination Btu per sec per sq in. All heat-flux rates will be given in these units unless otherwise specified. If the former units are more familar, it is simple to remember that one-million Btu per hr per sq ft is approximately equivalent to two Btu per sec per sq in.

Heat transfer in steam boilers and steam heat exchangers provides a large portion of the heat-transfer work done by mechanical engineers. In these applications heat-flux rates are as high as 85,000 Btu per hr per sq ft or approximately 0.17 Btu per sec per sq in. (1).[1]

Liquid-cooled internal-combustion engines have average heat-rejection rates of approximately 0.185 Btu per sec per sq in. This is representative of large aircraft engines of the 12-cylinder V type. Since heat transfer occurs during only 25 per cent of the cycle, intermittent peak heat fluxes are approximately 0.73 Btu per sec per sq in. (2).

Heat-flux rates in rocket engines vary with the type of propellant, size of combustion chamber, and operating pressures. Some representative

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Presented at a joint meeting of the AMERICAN ROCKET SOCIETY and the ASME, Los Angeles, Calif., June 29, 1949.

[1] Numbers in parentheses refer to Bibliography on page 77.

TABLE 1 REPRESENTATIVE VALUES OF AVERAGE HEAT-FLUX RATES OF DIFFERENT PROPELLANTS

Propellant designation		T_c, F	I_{sp}, sec	q_0, Btu/sec in^2
General	Specific example			
Low performance	Hydrogen peroxide	1050	110	..
Medium performance	Nitric acid Hydrocarbon	4000	192	1–2
Medium performance	Liquid oxygen Ethyl alcohol	4500	222	2–4
High performance	Hydrogen and oxygen	5500	310	3–8

heat-flux rates for liquid-propellant rocket motors are shown in Table 1. It is seen that, as the performance I_{sp} increases, the combustion temperature T_c increases and the average heat flux also increases. The heat flux for hydrogen-peroxide motors is not given because these motors can be operated indefinitely at thermal equilibrium without liquid cooling, and therefore the heat flux is not a problem in the thrust-chamber design. Unfortunately, the performance of hydrogen peroxide is so low, it is not a practical propellant by itself.

Conditions Affecting Heat Transfer in Rocket Engines: Before discussing various details, it might be well to review the phenomena taking place in a rocket combustion chamber. Two liquids are usually injected under high pressure into a combustion chamber. The liquids must be dispersed, vaporized, mixed, and burned. Combustion may take place in the liquid or in the vapor phase. The jets may intersect each other, or may splash on walls or special plates. The momentum of the jets after impingement may be axial, or it may have radial components. As the combustion proceeds, the composition of the products changes and the temperature increases. As soon as combustion is complete, which is desirable, the products, now at a calculable temperature and composition, are led into a De Laval nozzle where they accelerate, the temperature and pressure dropping as the velocity increases. With certain propellants the temperature and pressure condition are such that, because of

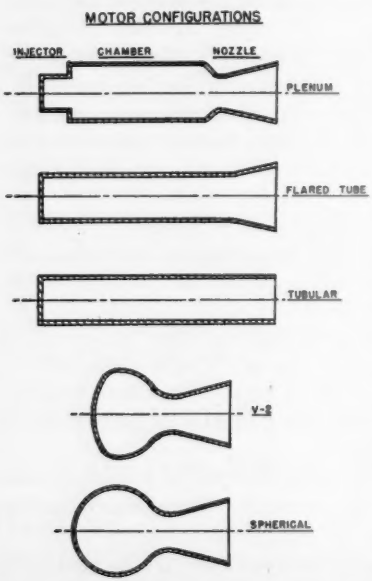

FIG. 1 MOTOR CONFIGURATIONS

TABLE 2 PROPERTIES OF ROCKET-MOTOR COOLANTS

Coolant	Specific gravity, 60 F	Freezing point, F	Boiling point, F	Specific heat, Btu/lb, F	Thermal conductivity, Btu/hr/ft,^{2} F/ft, 60 F	Viscosity, lb/ft/hr, 60 F	Stability	Remarks
RFNA	1.54	−54	129	0.47	0.112	2.74	Slight change in composition on long storage	Toxic vapors. Extremely corrosive
Aniline	1.03	21	364	0.49	0.10	12.61	Good	Toxic; absorbed by skin
Ethyl alcohol	0.792	−179	176	0.60	0.105	3.27	Good	Nontoxic in small quantities
Liquid O_2	1.14 (BP)	−362	−297	0.40	0.121 (−328 F)	0.458 (BP)	Stable but evaporates rapidly	Fire hazard
Hydrogen peroxide (90%)	1.40	21	285	0.62	.....	3.15	Poor; contaminates easily and then decomposes	Nontoxic; will cause fires easily
Hydrazine	1.01	34	236	0.745	.....	4.48	Poor	Toxic, dangerous, expensive
Liquid H_2	0.071 (BP)	−435	−424	2.25	.....	0.0335	Stable but evaporates very rapidly	Vapors highly inflammable over wide A/F ratio
Water	1.0	32	212	1.00	0.350	2.76	Excellent	Good coolant but bad propellant

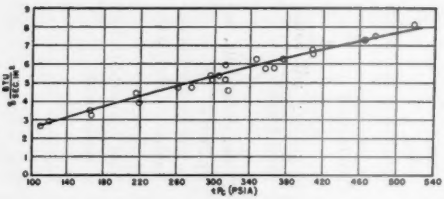

FIG. 2 VARIATION OF AVERAGE HEAT FLOW WITH CHAMBER PRESSURE—FLARED-TUBE CONFIGURATION—WATER COOLED

chemical equilibrium the combustion is incomplete. As the expansion continues, the lowered temperature changes the equilibrium constants, and the inhibited combustion proceeds with the liberation of more heat and a further change in composition. It is under these conditions of nonuniform combustion that heat-transfer calculations must be made.

Most reported heat-transfer data are based on tubes having a large l/d ratio. In liquid rocket motors this ratio rarely exceeds 4 to 1. Well-developed flow of a uniform nature is not present. Although circular cross sections are the rule in rocket motors now, the longitudinal shape may vary. Fig. 1 shows some of the shapes used in the design of combustion chambers. The flow through these odd shapes affects the heat transfer to some extent. The simpler looking shapes unfortunately are not used very much. In motors of large thrust, it is necessary to go to more nearly spherical chambers in order to reduce the tensile stresses.

Further complications arise from the necessity of operating at variable conditions of thrust, chamber pressure, and propellant flow. Fig. 2 shows the increase in heat flux with chamber pressure for a flared-tube type of motor. As chamber pressure and propellant flow are decreased, the heat flux also decreases, but not in direct proportion. Since only the propellant flow is usually available for cooling, the capacity for cooling decreases more rapidly than the heat loads. In addition, the coolant velocity and its film conductance decrease, tending to allow overheating of the metal walls.

The choice of coolants is restricted to the propellants and therefore substances normally unsuited as coolants must be used. Some of the fluids and their properties are tabulated in Table 2. These are certainly not materials one would voluntarily choose for efficient coolants.

Thermal Resistances: The total resistance to heat flow in a rocket combustion chamber is the sum of three individual resistances. These are: (1) Liquid-film resistance; (2) metal-wall resistance; and (3) hot-gas film resistance.

Without minimizing the problems and interest in the first two items, this paper will concern itself largely with the last item, namely, resistance of the hot-gas film, inasmuch as this presents the most perplexing problem in heat transfer in a rocket motor.

High-Energy Propellants—Hydrogen and Oxygen

Plenum Motor: A diagram of a plenum-style rocket motor utilizing a

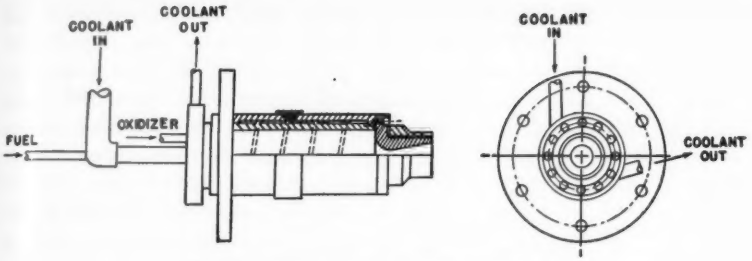

FIG. 3 LIQUID-COOLED MOTOR ASSEMBLY

helical cooling passage is shown in Fig. 3. This is strictly a test motor, being overly large and heavy.

The heat transfer to the motor shown in Fig. 3 is shown in Fig. 4. It is seen that the injector heat transfer is highest while the nozzle and chamber heat-transfer rates are somewhat less. The chamber, which has the largest surface area, fortunately has the lowest heat flux. The marked effect of mixture ratio on heat flow should be noted. The absolute value of heat flux is 3–8 Btu per sec per sq in., which is representative of high performance propellants.

Flared-Tube Motor: The heat flow to this motor as a function of chamber pressure was shown in Fig. 2. The distribution of heat flow as a function of chamber length is shown in Fig. 5. While the average heat flow is about 8 Btu per sec per sq in., the peak flow is approximately 15 Btu per sec per sq in., and this occurs forward of the throat. In this motor the velocity and temperature are extremely high forward of the throat. In the more conventional plenum motor the heat peak occurs at or immediately forward of the throat.

The accuracy of these heat-flow distribution measurements is not great, due to the heat contained in the boundary layer of the coolant fluid. A temperature traverse of the coolant passage is shown in Fig. 6. The thermal film thickness is approximately 0.010 in. This is for a water-flow velocity of 32 ft per sec or a Reynolds number of 85,000. The amount of energy contained in this boundary layer is remarkably high. In the investigation conducted with this motor, measurements were made of the metal wall temperature as well as of the coolant water temperature (at mid-passage depth). The results are shown in

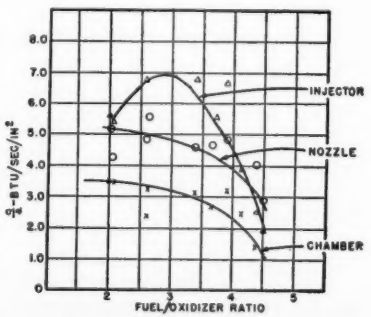

FIG. 4 HEAT TRANSFER FOR WATER CONVECTION COOLED MOTOR

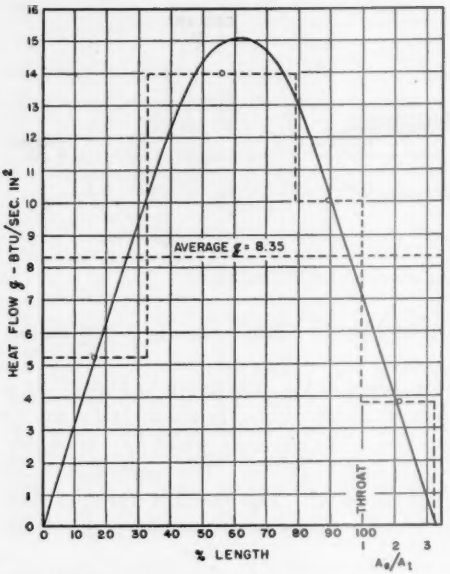

FIG. 5 HEAT TRANSFER FOR CONFIGURATION MOTOR, P_c, 300 PSIA, SIX-HOLE INJECTOR

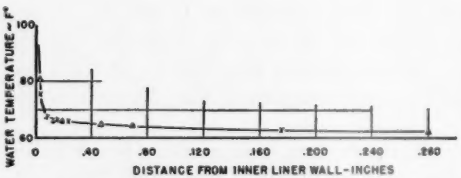

FIG. 6 TEMPERATURE DISTRIBUTION OF WATER PASSAGE AS MEASURED BY TRAVERSING THERMOCOUPLE

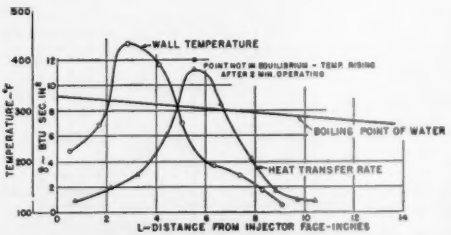
WALL TEMPERATURE
POINT NOT IN EQUILIBRIUM - TEMP. RISING AFTER 2 MIN. OPERATING
BOILING POINT OF WATER
HEAT TRANSFER RATE
TEMPERATURE - F°
BTU SEC IN^2
L - DISTANCE FROM INJECTOR FACE - INCHES

FIG. 7 COMPARISON OF MEASURED HEAT TRANSFER WITH CHAMBER WALL TEMPERATURE SHOWING EXPERIMENTAL INCONSISTENCY

Fig. 7. Coolant flow in this motor is from injector to nozzle. It is seen that the maximum measured heat flow occurs farther downstream than the maximum wall temperature. It is tentatively assumed that this represents the distance required for the boundary layer to transfer its energy to the bulk of the cooling water. It should also be noted that the wall temperature is 115 F greater than the boiling point of the water coolant. Boiling heat transfer undoubtedly occurred.

Another factor which complicates the heat-transfer problem is injector design. It is possible to operate a motor with different injectors and obtain the same chamber pressure, mass flows, essentially the same combustion products, flame temperature, and therefore the same Reynolds number. From these considerations one should expect the same film conductances and the same heat flows. Yet in Fig. 8 it is seen that doubling the number of fuel-injection holes reduces the heat flux 20 to 50 per cent. The effect of injection pattern on heat transfer is of paramount importance. This may be due to the different flame fronts created by different

injectors. Correlation of film conductances on theoretical grounds becomes of academic interest if the designer can change the heat flow 100 per cent by changes of injection pattern while keeping constant these factors which ostensibly determine the film conductance.

Medium Energy Propellants—Acid-Hydrocarbon: Motors for this propellant combination are usually of all-steel construction with axial or helical cooling coils. The heat transfer for a typical motor of this type is shown in Fig. 9. It is seen that the heat flux is much lower than for the high performance propellants and remarkably independent of mixture ratio. The gas film resistance ($1/hg$) is 2262 sq in. per sec F per Btu. The injector configuration is shown in Fig. 10.

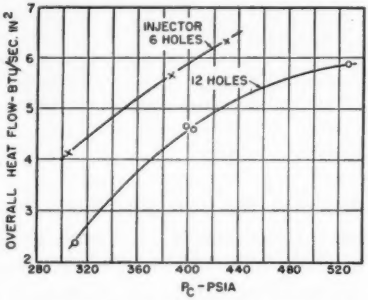

FIG. 8 OVER-ALL HEAT FLOW VERSUS P_c

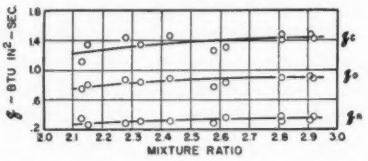

FIG. 9 HEAT TRANSFER WITH INJECTOR SHOWN IN FIG. 10
(q_c, chamber heat flux; q_n, nozzle heat flux; and q_o, over-all heat flux)

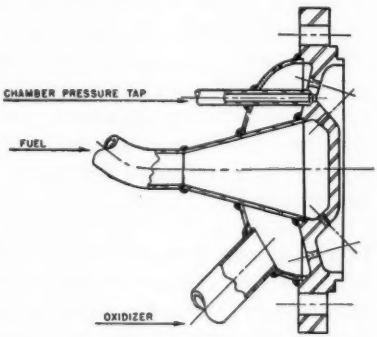

FIG. 10 INJECTOR PLATE

Heat-Transfer Calculations

Standard Equations: The discussion thus far has shown some of the severe problems facing the analyst in trying to calculate and predict the heat transfer in the combustion chamber of a rocket motor. The design, fabrication, and testing of liquid-propellant rocket motors are fairly straightforward. Heat-transfer data can be obtained from these tests. While the data are accurate to only ±10 per cent, they are reproduceable and valid. However, if rocket-motor design is ever to be more than an empirical art, it is necessary to attempt theoretical correlation of these heat-transfer data.

The first attempt is made, ignoring the complexity of the actual combustion phenomenon, by making the necessary assumptions of uniform distribution of the various fluid properties such as temperature, velocity, and composition. On this basis the classical equations for forced convection heat

TABLE 3 THEORETICAL COMBUSTION-GAS COMPOSITION

Nitric Acid—Hydrocarbon Propellant

T_c = 5021 F

Component	Mole Fraction	Component	Mole Fraction
H_2	0.0590	NO	0.0040
H_2O	0.3256	H	0.0089
CO	0.2191	N	0.0002
CO_2	0.1832	O	0.0018
N_2	0.1810	O_2	0.0054
C	0	...	..
OH	0.0199	...	..

transfer are applicable, if we assume fully developed flow.

Some of the classical equations are shown in the appendix. The first two equations utilize the analogy between heat transfer and fluid friction. This is particularly applicable for gases where the Prandtl number is fairly constant and is approximately unity. In addition the factor f is not critically dependent on an exact value of the Reynolds number due to the low value of its fractional exponent. This is fortunate in view of the uncertainty in obtaining the viscosity values required to calculate the Reynolds number. Equations [4], [5], and [6] are based on dimensional analysis and much empirical data, although Equation [6] (Reference 3) can also be derived from Equations [1] and [3] with a slight change in the coefficient. The use of Equation [5] (4) is difficult due to the usual lack of physical data, particularly κ and μ.

Physical Data: Physical data required for proper use of the foregoing equations, in general, consist of specific heat, c_p; density, ρ; viscosity, μ; and thermal conductivity, κ. There are adequate data for specific heat and density of all gases and gas mixtures over a wide temperature and pressure range. A particularly good source is (5). There are some low-temperature data on viscosity of gases, i.e., to 2000 F, but little on thermal conductivity. If viscosity at elevated temperatures can be estimated, it is possible to estimate κ by means of Prandtl's number which can be extrapolated fairly safely. A major difficulty is the calculation of viscosity and thermal conductivity for gaseous mixtures. The theory, while well developed (6), is quite complex and has not been reduced to practice.[2]

A theoretical combustion-gas analysis is shown in Table 3. The theoretical combustion temperature is also noted. The accuracy of calculated values of viscosity and thermal conductivity of such mixtures is extremely poor. Frankly, they are little more than educated guesses, probably to an accuracy of ±25 per cent.

Calculation of Film Coefficient: Over-all hot-gas film coefficients have been calculated from actual test data. This is not difficult since the heat flux is known and the water film and metal-wall resistance can be calculated fairly closely. Errors in these calculations are not serious since these two resistances are small compared to the hot-gas film resistance. Corrections

[2] Some pertinent papers on this subject, particularly by Hirschfelder, et al, have been published recently.

TABLE 4 THEORETICAL AND ACTUAL COMBUSTION-GAS FILM COEFFICIENTS

	Motor 1 (Fig. 3)		Motor 2 (not shown)	
	Bulk temp.	Film temp.	Bulk temp.	Film temp.
Reynolds' number	36,800	49,500	150,600	215,000
Prandtl's number	0.713	0.715	0.704	0.723
$f/2$	0.0029	0.0026		
Formulas	Film conductance, Btu/hr ft² F	Film resistance, sec in.² F/Btu	Film conductance, Btu/hr ft² F	Film resistance, sec in.² F/Btu
McAdams' simplified equation for gas	80	6460	...	...
Reynolds' analogy	94	5520	...	...
Von Kármán's extension of Reynolds' analogy	111	4680	...	...
Colburn equation (film)	138	3760	255	2040
Test data*	475	1091	771	672
Ratio $\frac{\text{Test data}}{\text{Colburn equation}}$	3.44		3.02	

CONTRIBUTIONS TO TOTAL HEAT RESISTANCE

Source	Resistance, sec in.² F/Btu	Per cent	Source	Resistance, sec in.² F/Btu	Per cent
Liquid film	68	4	Liquid film	62	8
3/16 - stainless steel (18–8) wall	542	32	Copper wall (9/32)	58	7
Gas film	1091	64	Gas film	672	85

* Corrected for radiation.

have been made for radiation in all of these calculations. This correction decreases the apparent film conductance. All data presented in Table 4 and Fig. 11 are based on average heat flux of cylindrical chambers only in plenum-style motors. Cases where boiling heat transfer occurred are not included. (Localized boiling heat transfer may have occurred unknowingly.)

Theoretical film resistances were also calculated, using all the equations of the appendix and the best possible estimates of physical properties at elevated temperature. Typical results are shown in Table 4. It is seen that while the Colburn equation gives the highest value of film conductance, this value is approximately one third of the measured value. The distribution of resistances between liquid film, motor wall, and hot-gas film is

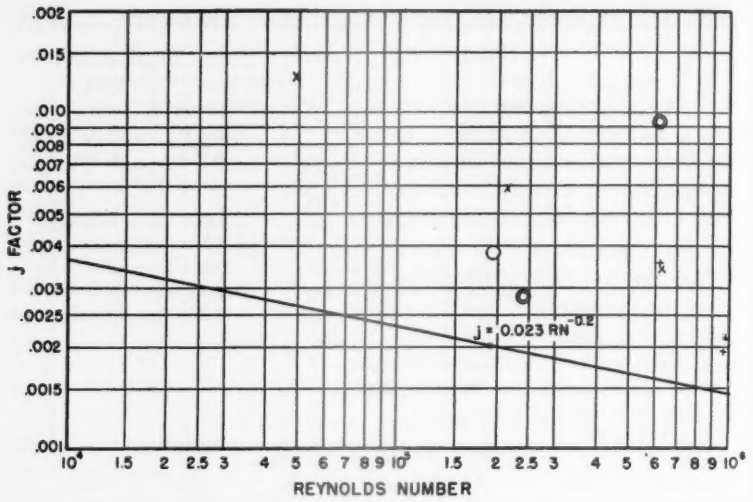

FIG. 11 HEAT TRANSFER FACTOR, j

[$j = (h/C_F G)(PrN)^{2/3}$. All physical properties are evaluated at average film temperature.]

also indicated. The hot-gas film has the largest resistance, especially in the case of the copper-walled motor.

A graphical comparison of theoretical and actual data is shown in Fig. 11. The film conductances are plotted as j factors which can be derived from the Colburn equation,

$$N\mu N = 0.023\ (ReN)^{0.8}\ (PrN)^{1/3}.$$

The conductance is much higher than predicted by the conventional equations. It is surprising that while the fluid properties are only "estimated," the data for radically different propellants and thrust chambers consistently fall higher than the predicted values. There are insufficient data to warrant deducing an equation.

Conclusions: Calculated theoretical film conductances do not agree with measured conductances. This may be caused entirely by the lack of suitable physical data. However there are many other possible explanations. Some of these are:

1 T_c is calculated but not measured. There may be large local variations in temperature, due to the combustion process.

2 Composition and velocity are not uniform and may vary widely. Due to the persistence of injection patterns through the combustion zone, there may be strong scrubbing of the chamber walls. Many motors exhibit a back-flow phenomenon in which the gases adjacent to the wall are found to be flowing from nozzle to injectors.

3 The possible lack of well-developed flow would result in smaller film thickness which could account for the higher than conventional conductivities.

The existence of combustion is felt to be a major disturbing feature. Some of the extremely high heat flows may be caused by free radicals or monatomic gases. Collisions between hydrogen atoms, for example, do not cause combustion unless a third body is present to absorb the energy of recombination. If a three-body collision occurs in the center of the chamber, this energy may be transferred to another atom or molecule, which however, passes it on to other hydrogen molecules causing dissociation and therefore no permanent recombination. If the atoms collide on the chamber wall, this energy is removed and the atoms recombine permanently. This preferential recombination on the chamber wall may well account for the extremely high heat rates observed with the high-performance propellants which always have an excess of hydrogen.

Techniques of Handling High Heat-Flux Rates

High Thermal Conductive Materials: There are fortunately many methods of handling these high heat-transfer rates. Some of the methods are applicable only to test-stand motors or to certain propellants. Some are universally applicable. One of the obvious methods is to use wall materials of high thermal conductivity. Copper, although one of the most desirable materials in this respect, is usually limited to small test-stand motors due to its excessive weight, poor corrosion resistance, and low strength.

The effect of material alone on wall temperature is shown in Fig. 12. This figure presents only a portion of the problem facing the designer. The use of any material must be based on a balance of the following factors: (1) Wall temperature; (2) strength of the material at the operating wall temperature; (3) thickness and weight of wall required for a given load; (4) corrosion resistance; and (5) modulus of elasticity at the operating wall temperature. The last factor is important because failure arising from instability of the inner liner under external pressure is a frequent cause of failure. Inasmuch as the liner is cooled by the propellant, the coolant pressure is greater than the combustion pressure by virtue of the injection drop. A particularly severe condition occurs at shutdown when full pump pressure exists in the cooling coils, the liner is still hot, and there is no counterbalancing chamber pressure.

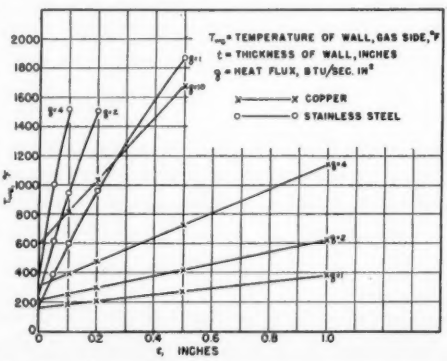

FIG. 12 COMBUSTION CHAMBER-WALL TEMPERATURE

Ceramic Insulation: Ceramic liners can be used to reduce the heat flux by interposing a thermal barrier.

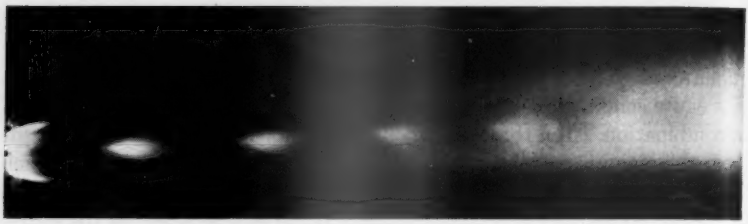

FIG. 13 EXHAUST FLAME FROM TRANSPIRATION-COOLED MOTOR

Thermal shock, bonding, retention of the ceramic lining, and weight are all practical problems to be solved in the use of this material. A requirement of any fabricational method is the ability to build motors of large thrust. While a ceramic-lined motor of the size of the A-4 is probably not impossible, its feasibility must be closely examined.

Film Cooling: Film cooling consists of injecting a small stream of a coolant, which may be one of the propellants, along the motor wall through small drilled holes. This places an artificial barrier between the hot gases and the motor wall. An A-4 motor utilizes this method of cooling as an adjunct to convection cooling. Each coolant stream protects a limited area of the motor wall. This is an extremely useful method.

Transpiration Cooling: An obvious extension of film cooling would be to use an infinite number of infinitely small holes to permit injection of the coolant over the entire combustion-wall surface. This method can be approached practically by the use of motor walls fabricated from porous sintered metals or ceramic materials. Theoretically this method is capable of protecting walls against any conceivable heat flux.

One of the many disadvantages of the last two methods is the fact that the portion of propellant used as a coolant does not mix completely with the main flow of hot gases. This represents a thermodynamic loss. A good example of this behavior can be seen in Fig. 13 which shows the exhaust flame of a transpiration-cooled motor. The existence of an unmixed protective sheath can be clearly seen.

Summary

Heat-transfer rates in rocket motors are extremely high. This may be due to: (1) Extremely high combustion temperatures and pressures; (2) flow with radial momentum; and (3) chemical action on the metal wall.

More important, the high heat flows cannot be accurately predicted from theory. This may be due to: (1) Lack of data on physical properties for the conditions of operation; (2) unknown conditions of temperature, flow, and composition, i.e., nonuniform flow; and (3) critical nature of injector design.

It is felt that the conventional heat-transfer formulas and theories are

not applicable to rocket-motor design. It will be necessary to have a detailed knowledge of the flow pattern inside a motor, the behavior of the combustion flame, and the injector pattern before heat flows can be predicted for new and unconventional rocket motors and propellants.

Bibliography

1 "Heat-Power Engineering," by W. N. Barnard, F. O. Ellenwood, and C. F. Hirshfeld, Part II, John Wiley and Sons, Inc., New York, N. Y., 1935.

2 Personal data, author.

3 "Heat Transmission," by W. H. McAdams, McGraw-Hill Book Company, Inc., New York, N. Y., 1942.

4 "A Method of Correlating Forced Convection Heat-Transfer Data and a Comparison with Fluid Friction," by A. P. Colburn, *Trans. American Institute Chemical Engineers*, vol. 29, pp. 174–210.

5 "Thermodynamic Properties of Propellant Gases," by J. O. Hirschfelder, F. F. McClure, C. F. Curtiss, and D. W. Osborne, OSRD Report No. 1087, November, 1942.

6 "The Mathematical Theory of Non-Uniform Gases," by S. Chapman and T. G. Cowling, Cambridge University Press, 1939.

APPENDIX

Heat-Transfer Formulas

A. Derived by the Analogy Between Heat Transfer and Momentum Transfer

Reynolds' analogy

$$\frac{h}{C_P G} = \frac{f}{2} \qquad [1]$$

This assumes $C_P\mu/\kappa \approx 1$, i.e., kinematic viscosity, μ/ρ = thermal diffusivity, $\kappa/\rho C_P$.

Von Kármán's modification and extension

$$\frac{h}{C_P G} = \frac{f/2}{1 + 5\sqrt{f/2}\left\{\frac{C_P\mu}{\kappa} - 1 + \ln\left[1 + \frac{5}{6}\left(\frac{C_P\mu}{\kappa} - 1\right)\right]\right\}} \qquad (2)$$

This does not assume $C_P\mu/\kappa = 1$ and takes into account the existence of laminar, buffer, and turbulent zones. This equation is valid for $C_P\mu/\kappa < 30$. To use these formulas it is necessary to obtain a value of f, the friction factor.

$$f = 0.046/Re^{0.2} \text{ for } 5000 < Re < 200{,}000 \qquad [3]$$

B. Derived by Dimensional Analysis

$$\frac{hD}{\kappa} = \alpha\left(\frac{DG}{\mu}, \frac{C_P\mu}{\kappa}\right) \qquad [4]$$

Nusselt number = function (Reynolds number, Prandtl number). Empirical tests have established the following constants:

$$\frac{hD}{\kappa} = 0.023 \left(\frac{DG}{\mu_f}\right)^{0.8} \left(\frac{C_P \mu_f}{\kappa}\right)^{1/3} \text{(Colburn)} \quad [5]$$

where μ_f = viscosity of the fluid at the mean film temperature, i.e., $t_f = (t_s + t_\infty)/2$.

If various simplifications are made, then for gases

$$h = 0.0144\, C_P \frac{G^{0.8}}{D^{0.2}} \text{ (McAdams)} \quad [6]$$

B. G. Nelson, Thiokol Corporation, Dies

IT IS with sincere sorrow that the American Rocket Society reports the death of Belton Glyn Nelson, chief chemical engineer of the Thiokol Corporation, Redstone Division, who died February 24, 1950.

Mr. Nelson, an active member of the American Rocket Society, was a native of Alabama and received his master's degree from the Alabama Polytechnic Institute in 1945. He was also a member of the American Chemical Society and the American Institute of Chemical Engineers.

Reaction Motors, Inc., Elects New Directors

DIRECTORS of Reaction Motors, Inc., Rockaway, N. J., recently announced that William A. M. Burden and Garrison Norton, partners in the firm of William A. M. Burden and Company, have been elected directors of Reaction Motors, Inc., one of the leading manufacturers of rocket engines for aircraft and guided missiles.

The William A. M. Burden and Company recently acquired a substantial interest in the rocket-engine firm, which will provide additional capital for the company and supplement the present interest of Laurance S. Rockefeller.

Reaction Motors, Inc., is a leading developer of rocket engines. Rocket engines are a new line of development in propulsion growing out of advances made in wartime. The Company pioneered in the development of the engines for the Bell X-1 and Douglas "Skyrocket" airplanes, and the power plants for the Martin "Viking" and Fairchild "Lark."

Mr. Burden, formerly Assistant Secretary of Commerce for Air, has been closely identified with aviation for several years. He was, during the war, a member of the National Advisory Committee for Aeronautics.

Mr. Norton, formerly Assistant Secretary of State, retired from the United States Navy with the rank of Captain after serving with distinction in naval aviation during the war.

FUNDAMENTAL PROBLEMS IN ROCKET RESEARCH

By Martin Summerfield[1]

Member ARS, Editor, Aeronautics Publication Program, Princeton University, Princeton, N. J.

IT IS frequently instructive to examine a field of research from a broad point of view in order to achieve a proper appreciation of its full scope and the foundations upon which it rests. In the field of rockets, technological progress has been made so rapidly during the past decade that there exists a tendency to overlook the fact that this progress was largely possible as a result of previous achievements in such fundamental sciences as chemistry, physics, and gas dynamics.

A direct illustration of this point is that it would not be possible today to conduct a rational development program in rockets if the theoretical methods of predicting values of specific impulse at all mixture ratios and pressures were not available. Yet, twenty years ago precise high-temperature calculations would have been impossible, for it was only during the decade of 1930–1940 that the application of statistical mechanics and spectroscopic data to the calculation of molecular thermodynamic properties took place. This thermodynamic research was made possible by the accumulation of spectroscopic information and the development of ideas of molecular structure that took place during the previous decade, the period 1920–1930.

The point is that fundamental research is an important part of rocket development, and that the possibilities for future progress depend in an essential way on the knowledge that is being sought at the present time in certain areas of basic science.

It is the purpose of this paper to illustrate this thesis by presenting various problems that have arisen in connection with the recent advances of rocket technology, and to indicate in some cases, the directions in which solutions might be found. In a brief survey of this type it is not possible to deal with more than a few topics, which admittedly were selected partly as a result of the personal interests of the author. Thus, no attempt has been made to deal with the problems to be overcome in the development of high-performance propellants [an interesting discussion of certain aspects of this field has been given by Zwicky (1a)[2]]. There will be no discussion of the high-temperature physical properties of materials, although it is

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[2] Numbers in parentheses refer to Bibliography on pages 96–98.

recognized that these are of great interest in the rocket field; nothing will be said about the problems involved in solid propellants; and many other interesting questions will be left untouched. The topics that have been selected for the present discussion are concerned with combustion, heat transfer, transport properties, and flow through exhaust nozzles.

Combustion Processes

Among the objectives of rocket-motor research are: (*a*) Reduction in the chamber volume necessary for complete release of the available heat of reaction, and simultaneously, (*b*) reduction in the intensity of heat transfer to the chamber walls. Both questions are fundamentally related to the space pattern of the combustion process in the chamber, which in turn is determined by the configuration of the injector, the velocities of the injected streams, the configuration of the combustion chamber, and, of course, the chemical kinetics of reaction of the propellants. The presently accepted method for making improvements in rocket motors is entirely empirical. Many different injectors are fabricated and tested and the best ones selected from the results of specific impulse and heat-flux measurements.

It is suggested that combustion-chamber development can be rationalized by (*a*) learning more about the chemical kinetics of reaction of the propellants, and (*b*) determining gas temperature and composition distributions within the chamber as functions of the configuration parameters. Knowledge of the kinetics of the reaction would lead to estimates of the minimum possible residence time, and hence the minimum volume required for complete combustion, and may lead to catalytic techniques for effecting even more rapid reaction. Temperature and composition surveys would reveal the degree of actual utilization of the available volume of a chamber, and lead to more effective designs.

The relationship between the kinetics of reaction and the volume of the chamber can be illustrated by consideration of the combustion of nitric acid and hydrocarbon fuel. It is reasonable to assume that, upon entering the chamber, the fuel is quickly decomposed by the intense heat and partial reaction with oxygen into simple gaseous molecules such as H_2, CO, CH_4, etc., and that the nitric acid is similarly decomposed to H_2O, NO_2, NO, N_2, O_2, etc., and that the principal combustion reaction takes place between these simple reactants. Of these, the most stable is probably nitric oxide, and since high combustion efficiency depends on the release of the oxygen in the nitric-oxide molecule either by decomposition or by reaction, it is reasonable to investigate the kinetics of nitric oxide as being an important rate-controlling step in the over-all process.

Extensive studies of the kinetics of decomposition of nitric oxide into oxygen and nitrogen, up to temperatures of 2500 K, carried out at the Uni-

versity of Wisconsin by F. Daniels and his collaborators, quoted in (1*b*), have indicated that the homogeneous-gas phase reaction is of second order, and that the rate constant in the forward direction is given by:

$$k_2 = 1 \times 10^9 \, e^{-(70{,}000 \text{ cal/RT})} \text{ atm}^{-1} \text{ sec}^{-1}$$

$$2 \text{ NO} \rightarrow \text{N}_2 + \text{O}_2$$

Although the conditions in the combustion chamber are much different from those under which the data were obtained, one may tentatively extrapolate them to these conditions (higher pressures, higher temperatures, and dilution with other chemically active gases), and thus estimate the residence time required for near equilibrium decomposition. Altman and Penner (1*b*) have carried out such calculations for various assumed temperatures in the neighborhood of the thermodynamic flame temperature. Their results are presented in Table 1, and the conversion of these results to values of L^* is given in Table 2.

Experiments with rocket motors burning nitric acid have indicated that L^* of the order of 50 in. is required for maximum specific impulse (2). Comparison with Table 2 indicates that the decomposition of nitric oxide may be responsible for this requirement since the actual temperature level in such motors is probably of the order of 2600 K. The conclusion that, in this particular case, a chemical process rather than a physical process such as evaporation of the fuel is the rate-limiting factor, is supported by the fact that, despite the large number and variety of injectors that have been tried, the minimum value of L^* of 50 in. has never been substantially reduced.

TABLE 1 RESIDENCE TIME TO REACH NEAR EQUILIBRIUM WITH RESPECT TO NITRIC OXIDE DECOMPOSITION AS A FUNCTION OF TEMPERATURE

Temperature, K	$t_c \times 10^4$, sec
2600	26.44
2700	13.96
2800	7.58
2960	3.06

From Reference 1*b*.

TABLE 2 MINIMUM REQUIRED L^* AS FUNCTION OF REACTION TEMPERATURE

T_c, K	L^*, in.
2600	56
2700	31
2800	17
2960	7.4

From Reference 1*b*.

These ideas should be followed up in a systematic manner, for they are of considerable significance in the field of combustion-chamber development. First, it is necessary to extend our knowledge of nitric-oxide kinetics to conditions more closely approximating those in a combustion chamber. Then, certain critical confirming tests should be made. One of these would be to test chambers of various amounts of heat rejection to the cooled walls, and observe the variation in minimum L^* with degree of cooling. Nitric-oxide decomposition is sensitive to temperature (Table 1), and if this analysis is correct, the L^* variation should show up strongly. Another test would be to inject prevaporized propellants. Unless fuel evaporation is an

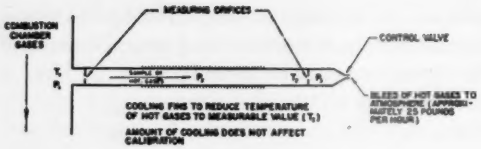
MEASURING ORIFICES
COMBUSTION CHAMBER GASES
CONTROL VALVE
COOLING FINS TO REDUCE TEMPERATURE OF HOT GASES TO MEASURABLE VALUE (T_2)
AMOUNT OF COOLING DOES NOT AFFECT CALIBRATION

FIG. 1 DOUBLE-ORIFICE TEMPERATURE MEASURING SYSTEM BASED ON STEADY WITHDRAWAL OF COMBUSTION GAS

important rate-limiting step, the effect on the minimum L^* should not be great.

Kinetics-research programs of this kind should prove valuable in clarifying the nature of the over-all process not only for nitric-acid systems but for other important propellants as well. One reaction that deserves more attention from kineticists than it is currently receiving is the decomposition and combustion of nitromethane. Nitromethane is still the best known propellant from the viewpoint of field handling, and investigations of its important characteristics should be encouraged (3).

It is to be hoped that in the investigation of the thermal decomposition kinetics of nitromethane the mechanism responsible for the thermal *detonation* of nitromethane will be discovered. Such detonations are found to originate in the cooling ducts of regeneratively cooled chambers despite the care that may be exercised in the design to avoid "hot spots." The hazard of detonation is the most serious drawback to the use of nitromethane at present.

It is essential to consider the problem of combustor design from an engineering point of view as well as from the standpoint of chemical reactions. An efficient design is one that approaches as closely as possible the limits set by chemical kinetics. To achieve this, it is necessary that the volume available in the chamber be effectively utilized for the active combustion process so that there be no unused dead regions, and so that all of the gas entering the nozzle attain the maximum flame temperature.

Temperature and composition surveys within rocket combustion chambers are the logical basis for such development. The instrumentation problems involved are obviously severe, but certain possible techniques should be explored. One instrument for measuring high gas temperatures, that has recently been described by D. W. Moore (4*a*), is based on the use of a cooled probe designed to permit a steady rate of gas withdrawal from the high-temperature zone. [See also the discussion of this method in Reference (4*b*).] It is illustrated schematically in Fig. 1. Two measuring orifices are involved: one located at the entrance or tip of the probe; the other downstream after a section designed to cool the gas to a moderate temperature level. The probe incorporates taps for measuring the pressure drop across the first orifice, and a thermocouple and pressure taps for metering the rate of gas flow through the second orifice. Knowing the mass flow through the system and the pressure drop across the first orifice, the temperature of the gas as it passes the first orifice can be calculated; and in

fact, the calculation can be accomplished mechanically or electrically so that the instrument can indicate temperature automatically.

The obvious advantage of this technique is that it is not necessary for the probe to come to temperature equilibrium with the hot gases, thus avoiding the limitations of other probes such as thermocouples and resistance elements. For many purposes it can be used more advantageously than optical temperature-measuring techniques, which generally are not adaptable to nonuniform temperature fields. However, significant errors may enter into the determination of extremely high temperatures with this probe. Thus, the entering gas may actually be precooled slightly while passing the first orifice, and the indicated temperature would be too low. Also, at rocket temperatures, the combustion gases contain a significant concentration of free radicals that would recombine in the tube between the first and second orifice. Such changes in the average molecular weight would tend to make the instrument read high. Finally, the method suffers from all the difficulties involved in obtaining a representative sample in a nonhomogeneous flowing gas mixture. The method nevertheless deserves serious investigation in connection with rocket development.

TABLE 3 RESULTS OF QUENCHING TEST WITH COOLED SAMPLING PROBE

Flame of Natural Gas with Oxygen
Oxygen/Fuel Ratio = 85% Stoichiometric
Pressure = 1 Atmosphere

Observed concentrations	Corresponding temperatures, F
$\frac{(CO_2)}{(CO)(O_2)^{1/2}} = 11.98$	4200
$\frac{(H_2O)}{(H_2)(O_2)^{1/2}} = 67.2$	4300

From Reference 5.

In considering the problem of sampling the gas composition in a combustion chamber, one of the important questions is whether the reaction can be quenched at the original composition. So far it does not seem possible to "freeze" the free radicals that exist at high temperature, although the mass spectrometer used in conjunction with a low-pressure ionization chamber is a possible means of detecting such free radicals. It may be possible, however, to develop a sampling probe with sufficiently intense cooling to freeze certain molecular reactions. An interesting set of experiments along this line has been reported recently by Erbe, Grey, and Beal (5). A flame of natural gas and oxygen at one-atmosphere pressure was sampled in the hottest region, and the composition of the sample was analyzed. The amounts of CO_2, CO, and O_2 were used to estimate the temperature at which the CO_2 recombination reaction was frozen. The amounts of H_2O, H_2, and O_2 were used to estimate the temperature at which the water-recombination reaction was frozen. Based on the results shown in Table 3, the CO_2 reaction seemed to freeze at 4200 F; the water reaction at 4300 F. Although the theoretical adiabatic flame temperature corresponding to the test-mixture ratio is several hundred degrees higher, the fact that the two derived values are so close together indicates that the actual local

temperature may have been of the order of 4300 F. Active research on gas-sampling problems of this kind is needed so that such techniques may be applied to combustor development.

Convective Heat Transfer in Liquids

Precise knowledge of cooling coefficients of liquids under a variety of conditions can provide the basis for a more accurate approach to the design of optimum cooling ducts in rocket motors. It is therefore of considerable importance to extend our knowledge of convective heat-transfer processes to cover ranges of heat flux that prevail in rocket motors.

An important question that arises in this connection is the generalization of the theory of convective heat transfer in fully developed turbulent flow to the case of nonuniform physical properties in the boundary layer. The presently accepted treatment of heat transfer between a surface and a turbulently flowing fluid is based on the analogy between heat transfer and momentum transfer, expressed by Reynolds in 1874, and later extended to Prandtl numbers different from unity by Taylor and von Kármán (6). In von Kármán's analysis the temperature distribution in the boundary layer is deduced from the velocity profiles observed by Nikuradse (7) in the flow of water in straight tubes. Since the velocity measurements were made under essentially isothermal conditions, the calculated heat-transfer coefficients rigorously apply only to the case of "isothermal" heat transfer, that is, to conditions of small heat flux.

The problem can be described by reference to Fig. 2, in which Nikuradse's data are plotted in the form of u^* versus y^*, the dimensionless mean velocity and distance from the wall, respectively (8). The data indicate the existence of three regions: (*a*) A laminar sublayer close to the wall in which viscous shear dominates turbulent shear; (*b*) a transition region

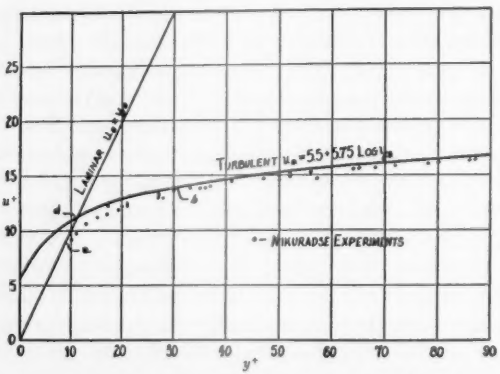

FIG. 2 VELOCITY PROFILE IN THE BOUNDARY LAYER ACCORDING TO NIKURADSE

in which viscous shear and turbulent shear are of similar magnitude; and (*c*) an outer region in which turbulent shear predominates. On the basis of the data shown in Fig. 2, von Kármán idealized the velocity distribution by setting the limit of the laminar sublayer at $y^*_1 = 5$ and the limit of the transition region at $y^*_2 = 30$. As shown in the figure, the velocity profile in the turbulent region follows a logarithmic relation. The laminar film is, of course, linear, and another logarithmic relation was fitted to the transition region. Examination of the theoretical expression for the heat-transfer coefficient discloses that it is quite sensitive to the velocity profile, in particular, to the value of y^*_1 and to the function for the transition region. In physical terms, the heat-transfer coefficient is sensitive to the thickness of the laminar film, and the structure of the transition region since these regions constitute the largest thermal resistance in the boundary layer.

Nothing is known about the behavior of these regions in nonisothermal flow, that is, under conditions of high heat flux. The liquid property that varies most strongly with the temperature variation in the boundary layer is the viscosity. The variation is such that the Prandtl number of aniline in a typical cooling duct of a rocket nozzle may be one tenth as much at the metal wall as in the middle of the duct. It is clear that the structure of that region in the boundary layer in which viscosity is important will depend on the viscosity gradient or on the gradient of the Prandtl number.

It does not seem possible to deduce the desired structure solely on theoretical grounds. It may be possible to derive a boundary-layer equation under these conditions by working backward from measured heat-transfer coefficients. An approach of this kind was attempted by Boelter, Martinelli, and Jonassen (9), but was unsuccessful, partly because the available data at high heat-flux densities were insufficient and contradictory, and possibly because the assumed form of the function was incorrect. More accurate data at high heat-flux densities, or specifically, with steep gradients of the Prandtl number in the boundary layer, are required in order to approach the problem in this manner. A beginning has been made on this task by Kreith and Summerfield (10, 11), and by McAdams and his colleagues (12). Data at more extreme viscosity gradients, however, are still required.

Direct experimental observation of liquid boundary layers, particularly in the region of the laminar sublayer should be considered despite the great difficulties involved. The region of interest is within a few thousandths of an inch of the wall. Clearly, unusual techniques are required for this task. Among those that suggest themselves are:

(*a*) Ultramicroscopic observation of the motions of colloidal particles, as employed by Fage (13).

(*b*) Interferometric measurement of the temperature distribution similar to the technique applied to the flow of gases by Kennard (14), Eckert (15) and Ladenburg (16).

(*c*) Optical beam-deflection measurements to determine the temperature distribution, as employed by Saunders, Fishenden, and Mansion (17) in the study of natural convection in liquids.

(*d*) Thermocouple-probe surveys with couples of the order of one mil thickness as in the investigation carried out by Sheppard (18).

Each of these methods has its limitations although each deserves to be reconsidered in terms of the problem at hand. An attempt was recently made by Thackrey and the author to make use of the birefringence exhibited by water containing one per cent of dissolved bentonite when subjected to viscous shear to determine the velocity distribution in the boundary layer. Although the results that were obtained are self-consistent, they are not in agreement with other known facts concerning boundary-layer flow, and it has been tentatively concluded that the method introduces certain errors of unknown character. The work will be published shortly.

Another important field for study is the process of cooling under conditions of surface boiling. It has been shown (10, 11, and 12) that when the heat flux is raised so that the surface temperature exceeds the boiling point of the liquid at the pressure existing in the duct, a bubbling action takes place at the surface, and further increases in heat flux are accompanied by only slight increases in wall temperatures. In effect, the boundary layer is more conductive in the boiling condition than in the nonboiling condition. At some limiting value of heat flux, however, the conductive character of the boiling film apparently breaks down and a burnout may occur (11). Close-up photographic studies of the bubbling action, at 3,000 frames per sec and at 20,000 frames per sec, have revealed that the

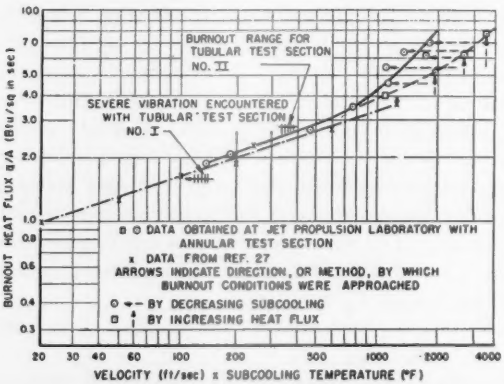
BURNOUT RANGE FOR TUBULAR TEST SECTION NO. II
SEVERE VIBRATION ENCOUNTERED WITH TUBULAR TEST SECTION NO. I
BURNOUT HEAT FLUX q/A (Btu/sq in sec)
100
7.0
5.0
4.0
3.0
2.0
1.0
0.8
0.6
0.4
0.3
⊡ ⊙ DATA OBTAINED AT JET PROPULSION LABORATORY WITH ANNULAR TEST SECTION
x DATA FROM REF. 27
ARROWS INDICATE DIRECTION, OR METHOD, BY WHICH BURNOUT CONDITIONS WERE APPROACHED
⊙ ↓ BY DECREASING SUBCOOLING
⊡ ↑ BY INCREASING HEAT FLUX
20 30 40 60 100 200 500 1000 2000 4000
VELOCITY (ft/sec) x SUBCOOLING TEMPERATURE (°F)

FIG. 3 VARIATION OF MAXIMUM HEAT FLUX, IN THE SURFACE BOILING REGIME, WITH VELOCITY AND DEGREE OF SUBCOOLING

transport of heat must be due almost entirely to the agitation in the boundary layer caused by the bubbling and only slightly to the physical removal of vapor containing the latent heat of vaporization (19).

At least one question is of importance to the engineer in connection with surface boiling. At what value of the wall temperature or of the heat flux does the heat-removal process break down, and how is the limiting value affected by the nature of the liquid, the nature of the surface, and the conditions in the system? It is known that certain regularities exist which provide a partial empirical answer to this question. Thus, Fig. 3 taken from (10) shows that the limiting heat flux increases with increasing velocity of the liquid and with greater subcooling of the liquid. (These data were taken with ordinary tap water.) No quantitative explanation is available for this behavior.

It is believed that a complete understanding of the process of heat transfer with surface boiling will be achieved only through a study of (*a*) the physics of nucleation in superheated liquids; (*b*) the dynamic behavior of bubbles in nonuniform temperature fields; and (*c*) the hydrodynamic effects in the boundary layer that accompany the rapid growth and collapse of bubbles. A start in this direction has been made by Gunther (19) at the Jet Propulsion Laboratory of the California Institute of Technology. By means of high-speed motion pictures it has been possible to trace the growth and collapse of typical bubbles. Fig. 4 illustrates the change in bubble radius with time superimposed upon an approximate temperature profile existing in the liquid adjacent to the surface. Despite the small dimensions of such bubbles (20 mils for those represented in the figure) they span a field of large temperature difference. Although some success has been achieved in the analysis of bubble motion in isothermal liquids as in the study of cavitation (20), the existence of a steep temperature gradient in the heat-transfer situation has complicated the problem severely.

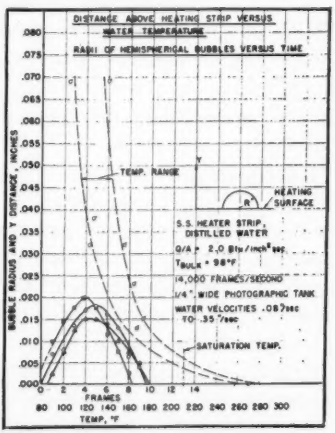

FIG. 4 GROWTH AND COLLAPSE OF BUBBLES DURING HEAT TRANSFER WITH SURFACE BOILING

Radiative Heat Transfer

An important question in connection with the general problem of heat exchange in a rocket motor is the amount of radiation emitted by the combustion gases in the chamber, and the manner in which the radiative heat transfer varies with pressure, temperature, and composition. Experimental emissivity data for the common combustion gases have been com-

piled by Hottel in (21). These data apply mainly to gas pressures of one atmosphere and extend only up to temperatures of 3600 F. Inasmuch as the conditions in a rocket motor involve pressures between 10 and 100 atm and temperatures possibly up to 8000 F, a re-examination of our knowledge of thermal radiation is required. The physical basis of radiation from gases is highly complicated, and in many cases the complexity of the spectrum is such that it seems impossible to carry out exact analyses. Certain general results, however, pertaining to the pressure and temperature dependence of the thermal radiation of gases can be derived.

Radiation from gases results from transitions between quantized states of the individual molecules of the gas. Reference to individual molecules in this discussion implies that the temperatures and densities are such that each molecule can be regarded as essentially uninfluenced by its neighbors. The number of transitions per unit time that occur between one energy state and another lower one varies with the number of molecules in the volume considered, the fraction of these molecules that are at any instant in the upper energy state, and the probability for the particular transition between the two states.

The intensity of radiation corresponding to a particular transition is therefore proportional to these factors, and to the extent that these factors are influenced by pressure and temperature, so is the intensity of radiation. In considering these effects an important question is whether the gas molecules are in statistical equilibrium with regard to the distribution over the available energy states. Radiation from gases in equilibrium is amenable to theoretical treatment; on the other hand, little is known about non-equilibrium radiation such as that emitted during the course of a chemical reaction or combustion process, sometimes called chemiluminescence. Some remarks will be made concerning chemiluminescence later. At this point the discussion will pertain only to radiation from gases in thermal equilibrium, or thermal radiation. Thus, in the case of a rocket motor, these considerations would apply to the conditions beyond the active combustion zone.

At the temperatures of rocket motors, the molecules of such gases as NO, CO, H_2O and CO_2 are almost all in the ground electronic state so that the intensities of electronic spectra are negligible in comparison with the intensities of the vibration-rotation spectra. In other words the bulk of the radiated energy lies in the infrared region. In the infrared spectrum of a gas one finds a number of bands and within each band an array of lines. The number of bands present generally increases with the number of independent modes of vibration of the molecule, which, in turn, generally increases with the complexity and asymmetry of the molecule. Thus, at room temperature the carbon-monoxide absorption spectrum exhibits only two bands, the fundamental at 4.7μ and the first overtone at 2.35μ while the spectrum of CO_2 exhibits at least ten bands of measurable intensity

(22). It may be readily appreciated that the analysis of the thermal radiation of CO_2 and H_2O will prove to be much more complicated than that of CO and NO.

The effect of pressure and temperature on band radiation depends upon several factors. As the temperature is raised, a larger number of bands become active as regions of appreciable emission, as a result of thermal excitation of larger numbers of molecules up to vibrational states of higher energy. Also, the individual bands become wider, their contours change, and more lines become visible as the statistical population distribution spreads to a larger number of rotational levels (23). The dependence of the emissivity of a gas upon temperature is therefore quite complex, but it has been possible to calculate this relationship for simple diatomic molecules.

A program of this kind was initiated by the author in 1948 while at the Jet Propulsion Laboratory of the California Institute of Technology. By an approximation method developed by Penner (24), these principles were applied to the calculation of the emissivity of carbon monoxide at one-atm pressure. The results are compared with the observations of Ullrich (25) in Table 4. It can be seen that the calculated emissivities are in reasonable agreement with the experimental values, for optical densities in excess of 2 ft-atm. The experimental data extend only up to 2500 F, much below the region of interest of the rocket engineer. On the strength of the agreement in the low-temperature range the calculations were carried out up to 5000 F. The most serious weakness in extending these computations to very high temperatures is that the available calculations of transition probabilities become less accurate as higher vibrational states become involved. At the basis of this uncertainty is the fact that the forces between the atoms of even such simple molecules as CO are only imperfectly understood. Nevertheless, in view of the great difficulty

TABLE 4 CALCULATED AND EXPERIMENTAL EMISSIVITIES FOR CARBON MONOXIDE AT VARIOUS TEMPERATURES AND OPTICAL DENSITIES

Temperature		pl = 6 ft-atm		pl = 2 ft-atm		pl = 0.2 ft-atm		pl = 0.05 ft-atm	
F	K	ϵ'_{calc}	ϵ'_{exp}	ϵ'_{calc}	ϵ'_{exp}	ϵ'_{calc}	ϵ'_{exp}	ϵ'_{calc}	ϵ'_{exp}
70	294	0.009	0.009	0.009	0.0062	0.009	0.0031	0.0085	0.0014
400	477	0.061	0.070	0.061	0.054	0.061	0.028	0.046	0.016
600	589	0.094	0.102	0.091	0.081	0.088	0.044	0.058	0.026
900	755	0.132	0.145	0.125	0.112	0.114	0.062	0.062	0.039
1200	922	0.151	0.155	0.139	0.123	0.114	0.070	0.053	0.046
1800	1255	0.138	0.115	0.119	0.093	0.079	0.056	0.030	0.037
2500	1644	0.106	0.086	0.090	0.070	0.048	0.042	0.016	0.029
3141	2000	0.077	..	0.064	..	0.029	..	0.0091	..
4041	2500	0.050	..	0.042	..	0.014	..	0.0043	..
4941	3000	0.034	..	0.028	..	0.0081	..	0.0025	..

From Reference 24.

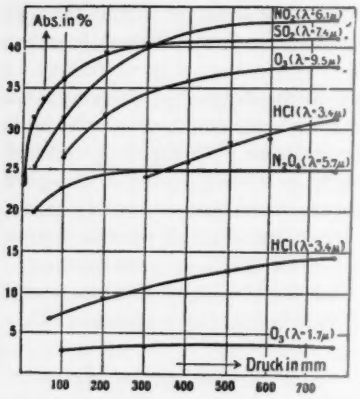

FIG. 5 EFFECT OF PRESSURE ON RADIATION ABSORPTION OF SEVERAL GASES

of performing such experiments at rocket temperatures it may be necessary for the engineer to rely on calculated emissivities, provided that the methods of calculation can be justified. Serious study, however, should be given to the extension of experimental data to high temperatures.

The effect of pressure on the radiative emissivity of a gas depends on the phenomenon of spectral line broadening (26). The effect can be demonstrated by determining the absorption due to a spectral band of an absorbing gas as the pressure in the optical cell is increased by the addition of a nonabsorbing gas such as nitrogen. Fig. 5, taken from E. von Bahr (27), shows that the absorption increases with increasing total pressure and at some "saturation pressure" reaches a maximum level value. In each case the cell contained a fixed mass of the absorbing gas at 300 K. It may be noticed that the saturation pressure of a heavy molecule tends to be lower than that of a light molecule. Thus from Fig. 5, the saturation pressures increase in the following order: N_2O_4, SO_2, CO_2, H_2O, HCl, and CO. Saturation occurs at about one-atm total pressure for CO_2 and H_2O, but exceeds five atm for CO. On the basis of the collision theory of line broadening, it may be expected that the saturation pressure will increase with increasing temperature, and that at rocket temperatures, the pressure-dependence of the emissivities of such gases as CO and NO, and possibly H_2O and CO_2, will be quite important, despite the high total pressure.

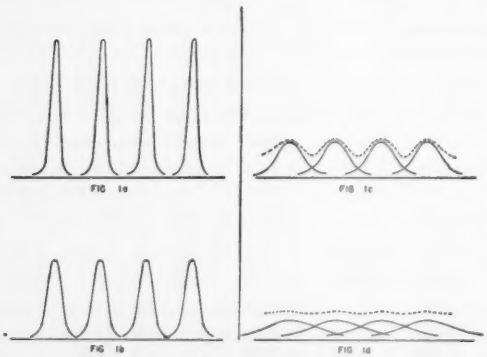

FIG. 6 PRESSURE BROADENING OF SPECTRAL LINES IN A VIBRATION-ROTATION BAND

The quantitative treatment of this departure from Beer's Absorption Law on the basis of the broadening of spectral lines has been considered by Elsasser (28) and studied experimentally by Summerfield (29) and by

Matheson (30). These considerations can be described by the aid of Fig. 6, which represents the intensity distribution of a series of rotational lines in an idealized band at several pressures. In (*a*), the total pressure is low, the lines are narrow, and the contours of the lines do not overlap appreciably. As the total pressure is increased, the lines overlap more and more as the individual contours become flatter, and finally in (*d*), the over-all intensity contour of the bands is quite flat. Although the integrated intensity of the band in condition (*d*) is equal to that in condition (*a*) in the range of pressures considered here, the energy emitted in the former case will exceed that in the latter case for finite thicknesses of radiating gas. Elsasser has shown this analytically. A physical argument is that a band in condition (*a*) will not radiate appreciably in the spectral regions between the lines and that even at infinite gas thickness the emissivity will be less than that of a black body. On the other hand, a band in condition (*d*) will radiate uniformly at all wave lengths within the band and will produce an emissivity of unity at infinite thickness. It may be deduced also from a consideration of Fig. 6 that the intensity of radiation will vary exponentially with gas thickness at high pressures, but in a more complicated way at low or intermediate pressures.

This discussion of thermal radiation may be summarized with the following points:

(*a*) In principle, it is possible to derive radiation emissivities of gases from spectral data although certain important questions remain to be answered.

(*b*) In the case of simple molecules the temperature dependence of the emissivity can be calculated and the results are in reasonable agreement with experimental data at large optical densities.

(*c*) The emissivities of gases do not in general follow Beer's Law; that is, the controlling parameter is not merely the product of the partial pressure and path thickness of the gas. Instead, the effects of pressure and thickness are generally more complicated and require consideration of line shapes and line broadening processes.

(*d*) Experimental data are required at temperatures between 3000 and 5000 F and at elevated pressures, although the author is not prepared to say how such experiments can be performed.

Any consideration of radiation in a rocket combustion chamber must include the chemiluminescent emission that may be present in the active reaction zone. It is not known whether such radiation exists to any appreciable extent in a rocket chamber. By the very nature of its non-equilibrium origin, it is not possible to predict either the spectral region or the intensity of such radiation without knowledge of the underlying mechanisms. Strong chemiluminescence has been observed by Kondratjew quoted in (31) in low-pressure carbon-monoxide flames, but it has also been observed that the addition of traces of water vapor or an increase in pressure

tend to quench such radiation. It is therefore possible that chemiluminescence may not be pronounced in a rocket combustion chamber.

It would be useful to measure the radiation intensity in various regions of a combustion chamber in order to determine whether appreciable chemiluminescence exists. A chamber having a greater length than the minimum necessary for complete combustion can be utilized to assure the existence of purely thermal radiation near the nozzle entrance, and thus, provide a check of the theoretical calculations; then, if the radiation intensity near the injector exceeds that near the nozzle, at least the difference can be ascribed to nonequilibrium mechanisms. Of course, the analysis of such radiation would also be of interest in connection with the task of identifying the reaction mechanisms that occur in propellant flames.

Transport Properties

An important aspect of the various fluid mechanical problems in rocket motors is the lack of adequate information on the transport properties, particularly viscosity and thermal conductivity, of the gases and liquids of interest in rockets.

In the interpretation of data on heat transfer to liquid fuels such as aniline, hydrazine, and nitromethane, precise knowledge of these properties is necessary, particularly in the temperature ranges encountered in cooling ducts. In general, these high-temperature data are not available, largely because such measurements in the past were conducted at atmospheric pressure, thus limiting the data to temperatures below the atmospheric boiling point. It is to be hoped that in the future the requirements of the engineer concerned with the high-pressure high-temperature cooling problems of rocket motors will receive some attention.

The transport properties of gases produced in rocket combustion processes are also of interest in connection with heat-transfer problems, and in this case, the data are desired all the way up to 5000 F and higher. Moreover, the actual problem is complicated by the fact that the combustion gas is a mixture of several gases of different molecular types. The need for such information arises in other researches as well, including the aerodynamic heating of high-speed vehicles, the diffusion of chain carrier atoms or radicals in flames, etc., and inasmuch as transport phenomena are closely related to the same intermolecular forces that determine the equation of state of a gas and other physicochemical properties, the securing of such data is a task of broad implications.

The state of affairs can be summarized as follows: for the common gases, viscosity data exist only up to about 1000 C, and thermal conductivity not above 300 C. For the main part the data were taken at one-atm pressure with the notable exception of steam for which there are data up to 290 atm. (32, 33). The consequence of this situation is that it has been

necessary to rely on extrapolation formulas, such as the temperature raised to 0.7 power law for the viscosity of air.

It is significant to note that real progress is being made in the theoretical calculation of viscosities of pure gases and gas mixtures over wide temperature ranges (34, 35). The method of analysis can be briefly described in the following way:

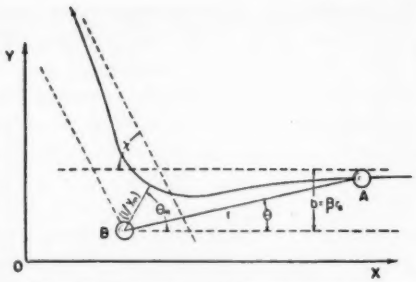

FIG. 7 PICTORIAL REPRESENTATION OF A NORMAL MOLECULAR ENCOUNTER, WITH LENNARD-JONES TYPE POTENTIAL

Viscosity is the macroscopic result of the transfer of momentum from rapidly moving molecules to slower molecules through the mechanism of intermolecular encounters. The results of these encounters can be treated statistically by the methods outlined in Chapman and Cowling (36) to arrive at the net transfer of momentum, and hence the viscosity. The heart of the problem is the calculation of the net effect of the statistical distribution of all possible collisions.

The kinematics of each intermolecular encounter is governed by the field of force that is assumed to exist between two molecules. The simplest idea is that the molecules bounce off one another like hard smooth billiard balls. Such collisions can be treated in a relatively simple manner with the result that the viscosity is proportional to the square root of the temperature, which is the formula of elementary kinetic theory. It is known that this formula is not quantitatively in agreement with the data.

Another possible model is that of attracting spheres, that is, the billiard balls are just as impenetrable as in the first case, but they attract each other with a force dependent on some inverse power of the separation distance. This model leads to Sutherland's formula, which is generally regarded as a fair approximation to the data over limited ranges of temperature.

Hirschfelder, et al (34), have adopted a Lennard-Jones type potential, according to which the molecule possesses a long-range attractive force plus a short-range powerful repulsive force. In this model the billiard ball is soft and an intermolecular encounter does not take the form of a sharp collision, but rather it results in an orbit not unlike that of a comet passing close to the sun (Fig. 7). The collision integrals in this case are complicated and laborious to evaluate, but they have been carried out and tabulated. Two parameters describing the potential field appear in these integrals, but they can be evaluated from the equation of state or from the available viscosity data at low temperatures. The results are such that, for example, in the case of air the calculated viscosities agree with the experimental values to within 2 per cent up to 1000 K and within 5 per cent

up to 1500 K. The integrals have been tabulated up to values that permit calculation of the viscosity up to 5000 K and higher if desired.

It seems that, as in the case of high-temperature thermodynamics, it may be possible to develop trustworthy theoretical values for high-temperature transport properties based upon accurate data at lower temperatures. But before this can be achieved it will be necessary to learn more about the nature of the intermolecular forces, and also about the extent of the transfer of energy to and from the internal degrees of freedom of a molecule during a collision. A step in this direction would be to obtain viscosity and conductivity data for pure gases and for mixtures over wide ranges of temperature for comparison with the theoretical results.

Flow Through Exhaust Nozzles

When very high performance rocket vehicles are considered it is generally found desirable to adjust the burning time of the propulsion system so that the burnout point will occur at an altitude of 100,000 ft or more. In other words, it is frequently desirable to traverse the dense region of the atmosphere at a moderate pace in order to minimize the aerodynamic loads and the aerodynamic heating problem, and to optimize the power-plant design. Under these conditions the ratio of the chamber pressure to external pressure would increase by a factor of 100 or more from the start to the end of powered flight. It is conventional practice to design the exhaust nozzle for some mean altitude, thus operating the nozzle in an overexpanded condition at sea level and an underexpanded condition at burnout.

It has been reported (37) that the jet detaches itself from the diverging cone when the area ratio of the nozzle is of such magnitude that the isentropic exit pressure would be equal to or less than approximately 0.4 of the surrounding pressure. In the reported investigation, which covered pressure ratios from 15:1 to 25:1 and divergence half-angles from 10 to 30 degrees, the separation behavior was remarkably reproducible and systematic. In order to facilitate further study of the phenomenon a series of similar experiments was performed in a small two-dimensional transparent exhaust nozzle using compressed nitrogen as the test gas (38).

Fig. 8 is a spark shadowgraph of the flow field during a typical test run. It was found in these experiments that the jet detached itself quite symmetrically as long as the pressure ratio exceeded about 10:1; that the ratio of the static pressure at the plane of separation to the external pressure was remarkably constant and equal to approximately 0.4; and that the pressure distribution along the nozzle wall followed the expected isentropic curve down to the point of separation and then returned sharply to atmospheric pressure.

This behavior is significant for its effect on the thrust produced by the rocket motor. All of that portion of the nozzle downstream of the plane

of separation adds nothing and subtracts nothing from the thrust, whereas without separation, the downstream portion would subtract from the thrust. In a powered flight to high altitude the resulting gain in total impulse acquired in this manner can be between 5 and 10 per cent depending on the operating conditions.

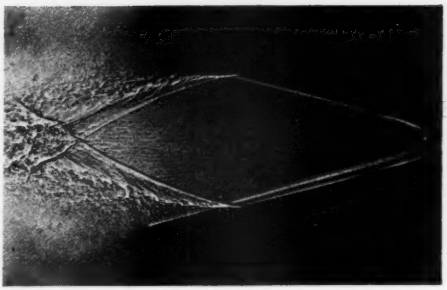

FIG. 8 FLOW SEPARATION IN THE DIVERGING PART OF AN OVEREXPANDED DE LAVAL NOZZLE

In effect, spontaneous jet separation is a step toward the ideal solution of a variable-area exhaust nozzle. The "exit area" adjusts itself automatically during the initial part of the flight, the separation plane moving gradually outward as the rocket rises. An additional gain in total impulse would be achieved if the jet could be induced to separate from the diverging cone not at the area at which the static pressure is 0.4 of the surrounding pressure, but at a position further upstream where the static pressure equals the surrounding pressure.

It has been shown by Weise (39) that the jet can be reattached by the application of suction through a slot in the wall located just downstream of the plane of spontaneous separation. Perhaps by the application of varying degrees of suction or pressure the separation plane can be controlled to the desired degree. In practice the suction or pressure at the slot can probably be obtained by connecting it either to the boattail or the nose of the rocket during flight.

Another practical application of controlled jet separation may be to apply steering forces to the vehicle during the initial period of powered flight by causing asymmetrical jet detachment through the appropriate use of pressure or suction at slots in the nozzle diverging cone. Whether sufficient asymmetry can be induced in this manner to develop the required forces can only be answered by experiment.

It might be remarked at this point that it is not understood why the jet separates in the manner shown in Fig. 8, nor have the controlling factors been identified. It is remarkable that the behavior in a two-dimensional channel with nitrogen is similar to that in a rocket nozzle, even to the precise area ratios at which separation occurs.

Conclusion

In conclusion it is apparent that involved in the field of rocket development are a large number and variety of fundamental problems, and that

the solutions to these problems are intimately connected with progress in the basic sciences. In fact, it can be said that this paper deals not so much with rockets as with chemical kinetics, combustion processes, molecular physics, spectroscopy, heat transfer, and fluid mechanics. The problems that were presented are, in many cases, logical extensions of the work that was carried out in these fields during several decades prior to the advent of the modern rocket motor.

However, the requirements of rocket development emphasize the need for extension of our knowledge in these fundamental fields, and indicate the necessity for a more critical examination of those properties of matter and those processes that are significant at extreme temperatures and pressures and at high rates. In this sense, these investigations constitute a logical part of rocket research.

Acknowledgment

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Bibliography

1*a* "Chemical Kinetics and Jet Propulsion," by F. Zwicky, *Chemical and Engineering News*, vol. 28, 1950, p. 156.

1*b* "Chemical Reaction and the Flow Process with Specific Application to Nitric Oxide Decomposition," by D. Altman and S. S. Penner, Progress Report 9-5, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Calif., 1947.

2 "Physics of Rockets," by H. S. Seifert, M. M. Mills, and M. Summerfield, *American Journal of Physics*, vol. 15, 1947, p. 121.

3 "Chemical Propellants—Mononitromethane," by F. Bellinger, H. B. Friedman, W. H. Bauer, J. W. Eastes, and W. C. Bull, *Industrial and Engineering Chemistry*, vol. 40, 1948, p. 1320.

4*a* "A Pneumatic Method for Measuring High-Temperature Gases," by D. W. Moore, Jr., *Aeronautical Engineering Review*, vol. 7, 1948, p. 30.

4*b* Article in "Temperature—Its Measurement and Control in Science and Industry," by H. F. Mullikan, 1941, Reinhold Publishing Company, New York, N. Y., p. 781.

5 "Study of the Use of a Cooled Probe for Sampling Hot Combustion Gas as a Means for Determining Heat Release," by P. E. Erbe, J. T. Grey, and J. L. Beal, Technical Memo. CAL-8, Project SQUID, Cornell Aeronautical Laboratory, Buffalo, N. Y., 1948.

6 "The Analogy between Fluid Friction and Heat Transfer," by T. von Kármán, *Transactions*, The American Society of Mechanical Engineers, vol. 61, 1939, p. 705.

7 "Laws of Turbulent Flow in Smooth Tubes," by J. Nikuradse, *Forschung auf dem Gebiete des Ingenieurwesens*, Verein deutscher Ingenieure, No. 356, 1932.

8 "The Mechanics of Turbulent Flow," by B. A. Bakhmeteff, Princeton University Press, Princeton, N. J., 1936.

9 "Remarks on the Analogy Between Heat Transfer and Momentum Transfer," by L. M. K. Boelter, R. C. Martinelli, and F. Jonassen, *Transactions*, The American Society of Mechanical Engineers, vol. 63, 1941, p. 447.

10 "Heat Transfer to Water at High Flux Densities With and Without Surface Boiling," by F. Kreith and M. Summerfield, *Transactions*, The American Society of Mechanical Engineers, vol. 71, 1949, p. 805.

11 "Pressure Drop and Convective Heat Transfer with Surface Boiling at High Heat Flux: Data for Aniline, N-Butyl Alcohol, and Water," by F. Krieth and M. Summerfield, Proceedings of Heat Transfer and Fluid Mechanics Institute, Berkeley, Calif., 1949, p. 127. (Published by The American Society of Mechanical Engineers, New York, N. Y.)

12 "Heat Transfer at High Rates to Water with Boiling," by W. H. McAdams, W. E. Kennel, C. S. Minden, R. Carl, P. M. Picornell, and J. E. Dew, *Industrial and Engineering Chemistry*, vol. 41, 1949, p. 1945.

13 "Turbulent Flow in a Circular Pipe," by A. Fage, *Philosophical Magazine*, vol. 7, no. 21, 1936, p. 80.

14 "Optical Method for Measuring Temperature Distribution and Heat Transfer," by R. B. Kennard, *Journal of Research*, National Bureau of Standards, vol. 8, 1932, p. 787.

15 "Interferometric Studies of Beginning Turbulence in Free and Forced Convection Boundary Layers on a Heated Plate," by E. Eckert, Proceedings of Heat Transfer and Fluid Mechanics Institute, Berkeley, Calif., 1949, p. 181. (Published by The American Society of Mechanical Engineers, New York, N. Y.)

16 "Interferometric Studies of Faster than Sound Phenomena. I. The Gas Flow Around Various Objects in a Free, Homogeneous, Supersonic Air Stream," by R. Ladenburg, J. Winckler, and C. C. VanVoorhis, *Physical Review*, vol. 73, 1948, p. 1359.

17 "Some Measurements of Convection by an Optical Method," by O. A. Saunders, M. Fishenden, and H. D. Mansion, *Engineering*, vol. 139, 1935, p. 483.

18 Thesis, by W. A. Sheppard, Department of Mechanical Engineering, University of California, Berkeley, Calif., 1946.

19 "Photographic Study of Bubble Formation in Heat Transfer to Subcooled Water," by F. C. Gunther and F. Kreith, Proceedings of Heat Transfer and Fluid Mechanics Institute, Berkeley, Calif., 1949, p. 113. (Published by The American Society of Mechanical Engineers, New York, N. Y.)

20 "The Dynamics of Cavitation Bubbles," by M. S. Plesset, *Journal of Applied Mechanics*, The American Society of Mechanical Engineers, vol. 16, 1949, p. 277.

21 "Heat Transmission," by W. H. McAdams, Chapter 4, 1942, McGraw-Hill Book Company, Inc., New York, N. Y.

22 "Das Ultrarote Spektrum," by C. Schaefer and F. Matossi, page 225, 1930, Julius Springer, Berlin.

23 "Molecular Spectra and Molecular Structure-Diatomic Molecules," by G. Herzberg, page 128, 1939, Prentice-Hall, Inc., New York, N. Y.

24 "The Emission of Radiation from Diatomic Gases: I, Approximate Calculations," by S. S. Penner, Progress Report 9–37, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Calif., 1949.

25 Thesis, by E. Ullrich, Department of Chemical Engineering, Massachusetts Institute of Technology, Cambridge, Mass., 1935.

26 "Pressure Effects on Spectral Lines," by H. Margenau and W. W. Watson, *Review of Modern Physics*, vol. 8, 1936, p. 22.

27 "Influence of Pressure on the Absorption of Infra-Red Rays by Gases," by E. von Bahr, *Annalen der Physik*, vol. 33, 1910, p. 585.

28 "Mean Absorption and Equivalent Absorption Coefficient of a Band Spectrum," by W. M. Elsasser, *Physical Review*, vol. 54, 1938, p. 126.

29 Thesis, by M. Summerfield, Department of Physics, California Institute of Technology, Pasadena, Calif., 1941.

30 "Intensity of Infra-Red Absorption Bands," by L. A. Matheson, *Physical Review*, vol. 40, 1932, p. 813.

31 "Explosion and Combustion Processes in Gases," by W. Jost, page 237, 1946, McGraw-Hill Book Company, Inc., New York, N. Y.

32 "Brief Review of Available Data on the Dynamic Viscosity and Thermal Conductivity for Twelve Gases," by G. A. Hawkins, *Transactions*, The American Society of Mechanical Engineers, vol. 70, 1948, p. 655.

33 "Experimental Determination of Heat Conductivity for Gases," by F. G. Keyes, *Transactions*, The American Society of Mechanical Engineers, vol. 71, 1949, p. 939.

34 "Viscosity and Other Physical Properties of Gases and Gas Mixtures," by J. O. Hirschfelder, R. B. Bird, and E. L. Spotz, *Transactions*, The American Society of Mechanical Engineers, vol. 71, 1949, p. 921.

35 "The Transport Properties for Nonpolar Gases," by J. O. Hirschfelder, R. B. Bird, and E. L. Spotz, *Journal of Chemical Physics*, vol. 16, 1948, p. 968.

36 "Mathematical Theory of Non-Uniform Gases," by S. Chapman and T. G. Cowling, Cambridge University Press, Cambridge, England, 1939.

37 "Flow Separation in Over-Expanded Supersonic Exhaust Nozzles," by M. Summerfield, C. R. Foster, and W. C. Swan. Paper presented at Heat Transfer and Fluid Mechanics Institute, Los Angeles, Calif., 1948. (Out of Print.)

38 Thesis, by J. D. McKenney, Department of Aeronautical Engineering, California Institute of Technology, Pasadena, Calif., 1949.

39 "The Separation of Flow Due to Compressibility Shock," by A. Weise, Tech. Berichte, Band 10, Heft 2, 1943. (Translated as NACA T. M. 1152, National Advisory Committee for Aeronautics, Washington, D. C., July, 1947.)

American Rocket Society News

Banquet Marks Official Launching of ARS Indiana Section at Purdue University

Two New Guggenheim Fellows Honored

ONE hundred American Rocket Society members and their guests attended the "Launching Banquet" held at the Purdue University Memorial Union Building in West Lafayette, Ind., June 8. The purpose of this banquet was to mark the completion of a successful organizing era and the beginning of the operating period for the Indiana Section—the newest section of the ARS.

Toastmaster C. M. Beighley, past-president of the Indiana Section and project engineer at the Purdue University Rocket Laboratory, opened the program by introducing the following incoming section officers: F. T. Rynes, president; C. R. Russell, vice-president; Gladys Geiger, secretary; and C. M. Ehresman, treasurer. These officers were elected at a meeting held May 16 and will serve for one year.

First speaker of the evening was William L. Gore, national president of the ARS and director of government operations for the Aerojet Engineering Corporation. After welcoming the new section into the national organization he went on to present an interesting history of the ARS. He concluded his speech by pointing out several possible peacetime uses of rockets

and expressing the hope that the use of rockets would not always be limited to military applications.

Dr. M. J. Zucrow, professor of gas turbines and jet propulsion at Purdue University and pioneer in the jet-propulsion field, introduced the principal speaker for the evening, Col. M. C. Demler, chief of the Power Plant Laboratory, Wright-Patterson Air Force Base. Colonel Demler's speech, "The Development of an Aircraft Rocket," outlined the engineering aspects of designing, fabricating, and testing a complete rocket-assist take-off unit such as is used on the B-36. Two movies which were used by Colonel Demler to illustrate his talk described the methods used by the Aerojet Engineering Corporation to develop JATO units, and the benefits which may be obtained by the use of JATO units on both light and heavy aircraft.

A third movie, "Operation Sandy," which was shown by President Gore, illustrated the launching of a V-2 rocket from the deck of the carrier *Midway*. Release of this film from the Navy classification list was obtained only three days prior to the showing at the banquet.

Dr. F. L. Hovde, president of Purdue University and formerly main chief of the Rocket Ordnance Division, Office of Scientific Research and Development, concluded the program by welcoming the new ARS section to Purdue University.

Engineers from several Indiana companies actively engaged in jet-propulsion research had been invited to attend the banquet. Among those engineers who accepted this invitation were 12 men from Haynes Stellite Corporation, Kokomo, Ind., and nine men from Allison Division of General Motors, Indianapolis, Ind. Visiting engineers inspected the Purdue Rocket Laboratory during the afternoon prior to the banquet.

As part of the dinner program, President Gore presented ARS pins to Eldon L. Knuth and John E. Scott, Jr., recently named as winners of Daniel and Florence Guggenheim Jet Propulsion Fellowships.

AT THE INDIANA SECTION BANQUET

Left to right: Eldon L. Knuth and John E. Scott, Jr. Members ARS, recent winners of Daniel and Florence Guggenheim Jet Propulsion Fellowships; W. L. Gore, President ARS

Story of these awards will be found on next page.

Interesting Spring Program

"Believing that an interesting program schedule is the key to success for an organization such as ours, I intend to concentrate my efforts while president on the obtainment of interesting speakers, movies, and inspection trips with which to bolster the members' enthusiasm," was the policy statement rendered by Clair M. Beighley when he was chosen to head the Indiana Section last February. Mr. Beighley has followed this policy.

On March 21, one month after the first organizational meeting, a constitution was presented to the prospective members of the Section for discussion and approval. With two minor amendments, this constitution was unanimously accepted and forwarded to the national office for the approval of the National Board of Directors.

On April 5, a 20-minute movie, "V-2 Rocket Assembly and Launching," provided the high light for a program attended by 150 rocket enthusiasts. An informative lecture on principles of rocket propulsion, illustrated with slides, was given prior to the showing of the film.

Next in this series of interesting programs was a lecture given on May 2 by W. L. Gilliland of Purdue University on "Rocket Fuels and Explosives." Dr.

DAVID H. ROSS, MEM. ARS, ONE OF THE RECENT WINNERS OF THE DANIEL AND FLORENCE GUGGENHEIM JET PROPULSION FELLOWSHIPS

Gilliland, a research chemist who has done much work on the development of fuels and explosives, discussed in an informative manner the fuels being used today as compared to the fuels which are available for use in future jet-propulsion devices.

Three ARS Members Awarded Guggenheim Jet Propulsion Fellowships

TEN young scientists have been granted Daniel and Florence Guggenheim Jet Propulsion Fellowships for advanced study in the field of rockets and jet-propulsion research and engineering. Three of these honored young men are members of the American Rocket Society.

John E. Scott, Jr., and Eldon L. Knuth are active in the Indiana Section of the Society, and David H. Ross is a member in Chicago.

John E. Scott, Jr., 22, a candidate for the degree of master of science in mechanical engineering and teaching assistant in Purdue University's School of Mechanical Engineering, completed his undergraduate training at the Virginia Polytechnic Institute in 1948, and served for one semester as a member of the VPI teaching staff.

David H. Ross, 22, is a member of the class of 1950 at Cornell University where he is a candidate for the bachelor degree in the Sibley School of Mechanical Engineering. Mr. Ross has been active in rocket developments at Cornell, and has been student assistant to Prof. P. F. Martinuzzi, professor of gas-turbine machinery at Cornell.

Mr. Scott and Mr. Ross will continue their work in rocket research at Guggenheim Jet Propulsion Center at Princeton University.

Eldon L. Knuth, secretary of the Indiana Section of the American Rocket Society, is a graduate of Purdue University and is now working for his MS degree in aeronautical engineering. He has done a great deal of work under Dr. M. J. Zucrow and will continue his work at the Daniel and Florence Guggenheim Jet Propulsion Center at the California Institute of Technology, Pasadena, Calif.

The American Rocket Society is honored to have three of its members earn this opportunity for advanced study.

Naval Air Rocket Test Station

ON APRIL 1, 1950, the Naval Ammunition Depot, Lake Denmark, Dover, N. J., was disestablished and the title of the Naval Aeronautical Rocket Laboratory was changed to Naval Air Rocket Test Station. The new station occupies the same physical facilities as formerly constituted the Naval Ammunition Depot, and is under the management control of the Navy's Bureau of Aeronautics.

The Naval Air Rocket Test Station (NARTS) is located adjacent to the Army's Picatinny Arsenal near Dover, N. J., and is about 35 miles west of New York, N. Y. Its primary mission is the static test and evaluation of rocket engines, their components and propellants. A small staff of well-qualified engineers and scientists is maintained for this work. In addition, specialized rocket-engine

test facilities are maintained for use by government contractors whose own test facilities may be, by virtue of their type, capacity, or location, inadequate or inappropriate.

Captain Roy Jackson, U.S.N., formerly overhaul and repair officer at the Naval Air Station, Seattle, Wash., is the first commanding officer of the new station. The technical staff includes several relatively young engineers whose names are already familiar to the rocket industry. Among these are Commanders Dayton A. Seiler, R. C. Truax and F. C. Durant III, of the Navy, and B. N. Abramson and Dr. John Clark.

New Books Available Through the ARS

PROPERTIES OF METALS AT ELEVATED TEMPERATURES, by George V. Smith. The book deals with the fundamentals of plastic flow and fracture of metals at ordinary and then at elevated temperatures, preparatory to a consideration of test apparatus and procedures used in evaluating metals for elevated-temperature service. 1950, 401 pages, $7.

HELICOPTER ENGINEERING, by Raymond A. Young. The book covers all approaches to helicopter engineering, including a complete section on jet helicopters and jet-driven rotors. 1949, 254 pages, $10.

SUPERSONIC AERODYNAMICS, by Edward R. C. Miles. The text is up to date in coverage of recent developments and includes some new advances, hitherto unpublished. The book tells the whole story with regard to each topic by the development in detail of the necessary mathematical concept. 1950, 255 pages, $4.

GERMAN-ENGLISH TECHNICAL DICTIONARY, compiled by U. S. Air Force, edited by Kurt F. Leidecker. Volume I, A–K. The Dictionary is the complete work of the U. S. Air Force in Europe handling the captured German material which was part of the German air power. The second volume of the dictionary will be ready for distribution in the fall. Volume 1, A–K, 494 pages, $17.50.

PRINCIPLES OF AIRCRAFT PROPULSION MACHINERY, by Israel Katz. A notable feature of this book is the effective balance between analytical and descriptive matter, and the integration of theory with practice. 1949, 477 pages, $6.50.

ARS Preprints Available

"The Application of White Fuming Nitric Acid and Jet-Engine Fuel (AN-F-58) as Rocket Propellants," by M. J. Zucrow and C. F. Warner.

"Estimated Performance of Hydrocarbon-White Fuming Nitric Acid Propellants," by M. J. Zucrow and C. H. Trent.

Price per copy: 25 cents to members; 50 cents, nonmembers.

The American Rocket Society

1949–1950

THE growth of the American Rocket Society from April, 1945, to April, 1950, has been consistent and all indications point to an acceleration of this growth in the next few years.

Membership

The membership in the ARS as of April 1, 1945 and April, 1950 shows the following comparison:

Year	1945	1950
Active	51	572
Associate	221	286
Student	46	144
Total	318	1002

These figures show an increase of 315 per cent over the five-year period.

One interesting development over this period was the drop in the associate and student members during 1948–1949. In September, 1948, the annual dues for associate and student members were increased. This increase resulted in a distinct drop in the associate group—those whose interest in rockets and jet propulsion was limited—and a slight drop in the student roster (due without a doubt to a purely financial situation). The associate group since 1949 has decreased but at the same time the increase in active membership has more than offset the associate drop. The student group leveled off after the increase in dues and is now definitely increasing.

Corporate and Affiliate Corporate Members

In 1948 the ARS instituted a campaign to secure corporate members with annual dues of $250. The following companies now have Corporate memberships in the Society.

Aerojet Engineering Corporation
Haynes Stellite Corporation
Reaction Motors, Inc.
Laurence Rockefeller
Thiokol Corporation
United Aircraft Corporation
Curtiss Wright Propeller Division

In 1949 the ARS instituted a campaign to secure affiliate corporate members (a combination of advertising space in the JOURNAL and an associate membership ranging from $100 to $400 per year). The Society now has the following affiliate corporate members:

Louis J. Curran
Louis M. Barish & Company
Sealol Corporation
Glenn L. Martin Company
Hycon Corporation
Tubing Appliance Company
Miniature Precision Bearings

Sections

Within the last two years the ARS has established several sections throughout the United States. ARS membership is broken down by Sections as shown in Table 1.

Subscribers

Subscriptions to the JOURNAL of the ARS are limited to libraries and research organizations. Individual subscriptions are not accepted since the JOURNAL is part of the individual membership service. An analysis of the five-year period shows the following comparison:

Year	1945	1950
Subscriptions	60	394

An interesting fact is that the JOURNAL of the ARS is sent to libraries in 17 foreign countries including the new nation of Israel.

Financial Report

The financial report for the ARS covers the years 1948, 1949, and the first four months of 1950. An analysis of the assets and liabilities of the Society appear in Table 2.

An analysis of the income and expenditures of the ARS for the years 1948, 1949, and 1950 appear in Table 3.

TABLE 1 ARS MEMBERSHIP BY SECTIONS

	Active	Associate	Student	Total
New York Section	265	90	45	400
Washington, D. C.	25	25	7	55
Southern California	114	19	12	145
New Mexico–West Texas	58	16	2	76
Indiana	4	3	23	30
At Large	108	133	55	296
Total	572	286	144	1002

TABLE 2 ANALYSIS OF ASSETS AND LIABILITIES

	Assets	Liabilities	Net Worth
1947–1948	$ 9,402.87	$ 914.67	$ 9,458.20
1949–1950	11,673.27	2,497.44	8,175.83
April, 1950	15,644.08	1,374.84	14,269.24

TABLE 3 ANALYSIS OF INCOME AND EXPENDITURES

	1948	1949	April, 1950
Income	$7,198.80	$12,164.73	$5,430.14
Expenditures	7,169.42	11,814.25	5,090.06
Net Balance	$ 84.87	$ 350.48	$ 340.08

ARS St. Louis Session

THE American Rocket Society sponsored a technical section at the 1950 semiannual meeting of The American Society of Mechanical Engineers, held in St. Louis, Mo., June 19–23, 1950.

About 50 members and guests heard two papers presented. The first was "Estimated Performance of Hydrocarbon-White Fuming Nitric Acid Propellent," by M. J. Zucrow, professor of gas turbines and jet propulsion, and C. H. Trent, department of engineering, Purdue University, West Lafayette, Ind.; and the other was "The Application of White Fuming Nitric Acid and Jet-Engine Fuel as Rocket Propellant," by Dr. Zucrow and C. F. Warner, department of engineering, Purdue University.

Dr. Zucrow was unable to attend the meeting so the papers were read by Mr. Trent and Mr. Warner.

In his summary Mr. Trent said that "specific impulse based on frozen equilibrium of several pure hydrocarbons oxidized by WFNA has been calculated at combustion pressures ranging from 200 to 2058 psia. In addition, the effects on specific impulse characteristic velocity, and thrust coefficient caused by a variation in C/H ratio of the fuel, O/F mixture ratio, and combustion pressure have been considered. Since the C/H ratio of the fuel exercises only a small effect on the specific impulse, the performance of saturated hydrocarbons of C/H ratio of 5.25 to 5.71 can be represented by that for octane (C/H = 5.33).

"At all of the combustion pressures studies, the maximum values of specific impulse were obtained at mixture ratios slightly rich in fuel; that mixture ratio producing gases having a lower mean molecular weight. A significant increase in the specific impulse at any mixture ratio results by increasing the combustion pressure," he said.

In recent years, according to Mr. Trent, there has been a great interest both in the United States and in certain European countries in the application of white fuming nitric acid (WFNA) and a hydrocarbon fuel as rocket propellants. In this country jet-engine fuel AN-F-58 has been selected as the hydrocarbon fuel. The principal reasons for the selection of these propellants are apparently related to the problems of availability and logistics.

"Both propellants are liquids, can be made readily available in an emergency, are relatively inexpensive, and have satisfactory freezing points. Furthermore, the performance of the WFNA-jet-engine fuel propellants is not greatly different from the nitric acid-aniline or aniline-furfuryl alcohol propellants with which a large body of experience has been accumulated.

"The fields of application of the WFNA-jet-engine fuel propellants are envisioned as being those where the consumption of propellants will be large, and the operating requirements can be satisfied with propellants giving performances substantially equal to those obtainable with the nitric acid-aniline type propellants," he reported.

Southern California Section

THE Southern California Section of the American Rocket Society ended an active 1949–1950 season on June 28, 1950, with a dinner meeting held at the Pasadena Athletic Club. Two papers were presented.

The first was "Statistical Methods Applied to the Misalignment of Rocket Jets" presented by A. Stanley of Hughes Aircraft, and the second was "Problems in Instrumentation of Guided Missiles," presented by G. M. Giannini and R. Powell of G. M. Giannini Company. The meeting was attended by approximately 65 members and guests. Some of the guests present included personnel from Wright Field, University of Michigan, and Purdue University.

According to R. B. Canright, secretary-treasurer, the Section is looking forward to an interesting program of meetings this fall.

SUPERSONIC AERODYNAMICS

A Theoretical Introduction

By Edward R. C. Miles

Research Mathematician, Institute of Cooperative Research

255 pages, 106 illustrations, $4.00

Here is a timely new book, just published, that assembles information needed for further theoretic investigations of supersonic aerodynamics. Emphasizing the mathematical approach, it develops necessary mathematical ideas in detail, and presents modern data on the nature of characteristic curves, in two or three dimensions, and in semi-empirical formulas for the drag coefficient and center of pressure of ogives. Every important phase of the subject from thermodynamic preliminaries and the divergence theorem, to the Taylor-Maccoll method for cones in supersonic flow, is carefully covered.

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- **emphasizes the necessary mathematical concepts**
- **describes fully the nature of characteristic curves**

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